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Icing of Carburetor Air Induction Systems of Airplanes and Engines

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SUMMARY

In conjunction with the development of a new commercial airline powerplant installation, laboratory tests were made of the icing of all critical parts of the carburetor air induction system, from the air scoop entrance to the supercharger impeller. Flight tests were made to establish a basis for the laboratory tests. The laboratory program and apparatus were examined by all concerned and the results obtained were compared with actual operating experience to insure that they had a direct practical application. Laboratory tests included about 75 runs of full scale icing of the scoop in an 18 × 36 in. refrigerated wind tunnel and about 275 full scale runs of carburetor and engine induction system icing with a selected fuel in a complete running engine rear section. A laboratory was constructed for the latter tests, which included refrigeration, heating, and humidification equipment for producing cold and preheated carburetor air over a wide range of conditions.

The results of this investigation were: (1) a recommendation for radical changes in the original design of an air scoop which should make it possible to operate the scoop, without danger, under the most severe impact icing conditions; (2) the effect of a wide range of a large number of variables on the formation of ice on all parts of the induction system; (3) the extent of the induction system icing hazards, and effectiveness of preheat in preventing and removing ice formed with a conventional type and a new type of fuel discharge nozzle; (4) the relative effectiveness of preheat and alcohol in removing induction system ice.

INTRODUCTION

AS A PART of the development of a new powerplant for a commercial airline installation, Pratt & Whitney Aircraft undertook the task of evaluating the hazards incident to icing of the engine parts of the carburetor air induction system. In spite of the magnitude of this program, tests were also made in an effort to eliminate icing hazards of other parts of that induction system, so that full advantage could be taken of the better engine performance obtained, resulting in better operation of the airplane. The object of this study was to obtain a non-icing engine induction system, and to provide a foundation of basic data for this and future developments of induction system de-icing.

The major parts of the induction system tested were a scoop, scoop adapter, throttle body, fuel nozzle adapter and blower throat. The scoop included a cold air entrance, an auxiliary entrance for preheated air, a preheater valve for controlling flow of the hot and cold air, an elbow for changing the direction of the air through an angle of 107°, and a straight section 3 in. long. The straight scoop adapter was 7 in. long, flexibly connected to the scoop and bolted to the throttle body. The throttle body was a double barreled type with two throttle butterflies and metering elements for automatically controlling fuel to air ratio. The fuel nozzle adapter was about 1 in. long and supported one type of fuel discharge nozzle with its X-bar fuel distributor. The direction of air flow was changed from almost vertical to horizontal in the blower throat which included a more recent design of fuel nozzle, hereafter called new fuel nozzle.

Tests were made with full scale equipment including a complete running engine rear section for designing a satisfactory carburetor air scoop and preheater valve for operation in snow or sleet, and to develop a satisfactory type of fuel discharge nozzle which would eliminate icing of the engine part of the induction system. However, before such investigations could be planned intelligently, it was necessary to make a number of tests to establish the basic effect of the important variables on the formation of ice because the general information available on the subject was inadequate. At the time this investigation was initiated, there were no adequate air-conditioning facilities for carburetor air available for this study, consequently it was necessary to construct a complete test plant for the engine part of the investigation. The air scoop icing tests were made in an 18 × 36 in. refrigerated wind tunnel. The fundamental part of the investigation was completed for both an X-bar fuel distributor and the new fuel

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nozzle, including prevention and removal of ice with preheat and a comparison of the effectiveness of removal of ice with alcohol and preheat. Conferences were held with military and commercial airplane operators and with others so that their most recent experience could be utilized in planning the study, in designing the test equipment, in prescribing the method of test and in interpreting the results. Because of the completeness of the equipment, the magnitude of the testing and the accuracy of control and measurement of the known variables, the application by the industry of the information obtained from this investigation should eliminate all known hazards of icing of the engine parts of the induction system tested. It should further help to eliminate the hazards of icing of future systems if the information presented herewith is utilized in their design. More development work should be done on the scoop and carburetor.

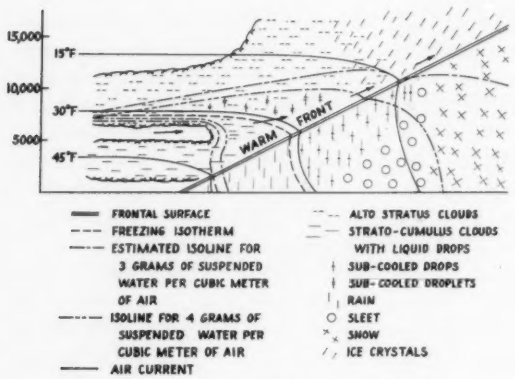

FIG. 1. Typical atmospheric conditions in over-riding air masses.

Airlines have flown a total of 287 million miles during the last three years with the problem of induction system icing continually before their pilots. Despite that fact, almost no complete data on the history of icing incidents during operation are available in airline records. The only practical way of investigating the wide range of atmospheric conditions encountered during many years of operation is to conduct laboratory tests. The laboratory also has the advantage that more precise control and measurements are possible and the system may be quickly dismantled to observe ice which has formed. There is always some doubt when laboratory tests are extrapolated to actual operation, but any difficulty will be eliminated if the results are checked at a few similar operating conditions.

The study was divided into two main parts, first, the icing of the scoop at The B. F. Goodrich Company and, second, the icing of the carburetor and engine parts of the induction system at the York Ice Machinery Corporation. In connection with the latter, flight tests were made before the laboratory investigation was started to insure that the effect of uncontrolled variables on induction passage metal temperatures of the engine were unimportant.

Shortly before this study was started, an N.A.C.A. Special Sub-committee of the Committee for Power Plants for Aircraft was organized to investigate the subject problem in general and more particularly to find immediate solutions for existing military and airline induction system icing problems. The study presented herewith has been coordinated with the N.A.C.A. work through the author who is a member of the committee. It supplements, rather than overlaps, the N.A.C.A. program, because the method of test differs and no work has been done by them with the new fuel nozzle.

Analysis of Operating Conditions

The variables of induction system icing may be classified into those of the atmosphere and those of the airplane. In this investigation, the induction systems tested did not include much of the airplane installation. The effect of all parts of it on icing of the engine part of the induction system is probably small except, perhaps, for rear engine compartment cooling air temperature. Effect of its variation on icing was estimated by heat transfer calculations and found to be small. Only atmospheric conditions and engine operating conditions were therefore controlled at York. Unfortunately such an accurate simulation of laboratory conditions with flight could not be obtained in the scoop icing part of the study.

Atmospheric Conditions

The atmospheric variables which have the most effect on induction system icing are air temperature and water content. Unlike other airplane icing problems, ice may form in the carburetor or engine intake at air temperatures considerably above 32° F. with relative humidities greater or less than 100 per cent. The atmospheric conditions of most interest in this study are those where the water content is high, that is, near to or greater than 100 per cent relative humidity. Dangerous excess icing water exists in the atmosphere in the form of clouds, fog, rain, snow, sleet, supersaturated vapor, and sub-cooled liquid. An example of a common U.S. winter weather condition in which these occur is illustrated in Fig. 1. The condition is produced when warm moist air of a tropical origin overruns cold dry air of a polar origin. The formation of induction system ice is possible both above and below the warm front. If possible pilots avoid such an atmospheric condition in operation.

Engine Operating Conditions

The following are a few miscellaneous comments supplementing Table 1 which summarizes normal engine operating conditions:

1. Power output controls carburetor air flow and b.m.e.p. controls throttle position. For future reference, carburetor air temperature is defined as the average air temperature at the inlet to the carburetor.

TABLE 1

Typical Engine Operating Conditions

Operating Condition	% Rated Power	% Rated R.P.M.	B.M.E.P. lbs./sq. in.	F/A
Ground running	2–60	3–80	12–120	.075–.090
Take-off	85–100	90–100	175–195	.105
Climb	70–75	80–85	156–160	.080–.090
Cruise	52–67	80–85	116–140	.068–.072
Approach for landing	25–30	55–70	70–80	.075–.085

2. There are very few incidents on record where icing difficulties were encountered in take-off or climb, if proper precautions were taken to prevent ice from forming during ground operations.

3. In airline operation, with Pratt & Whitney installations, preheated air is used only about 5 per cent of the total operating time.

4. The installations use a carburetor preheater capable of producing a 110° F. preheated carburetor air temperature at a strut temperature of 0°F. and cruising power.

5. At higher strut temperatures, the preheater is capable of producing higher carburetor air temperatures, for example, at 60°F. strut temperature such a preheater could produce a carburetor air temperature greater than 170°F. in the full-hot position. Continuous operation at such a high carburetor air temperature is not desirable, because there is a possibility of detonation. However, momentary use of the full-hot position even under those conditions is recommended if dangerous ice has been allowed to accumulate. Because of the possibility of detonation, for continuous operation with high preheat capacity at high strut temperatures, it is desirable to obtain satisfactory operation with partial preheat.

6. Besides its effect on detonation, the only important effect on available engine power of increasing carburetor air temperature is to reduce that power above critical altitude, since at lower levels all operation is at part throttle. The effect at critical altitude is only important in the case of an emergency such as single engine operation over mountains.

Summarizing, some type of engine induction system ice may occur with any engine operating condition, but take-off and approach or glide are the two most hazardous. At take-off even minor engine operating difficulties cannot be tolerated. During approach or glide ice forms more readily and a larger loss in available power may occur before it is detected.

Thermodynamic Description of Induction Processes

The six most important positions in an induction system from a thermodynamic point of view are indicated in Fig. 2. They are: (1) free air, (2) scoop entrance, (3) entrance to the carburetor, (4) flow restriction at the throttle, (5) fuel nozzle, (6) blower throat. From

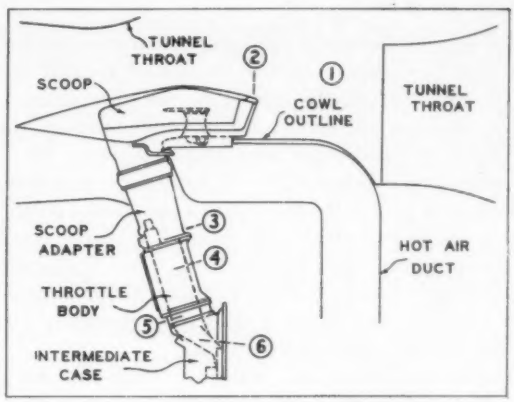

FIG. 2. Wind tunnel test installation.

(1) to (2), the air is compressed adiabatically to a higher pressure and temperature by the ram effect of the airplane velocity. The full adiabatic pressure rise at 150 and 450 m.p.h. are 11 and 109 in. water at sea level and the temperature rises at the same speeds are 1.7°F. and 36°F. The pressure change from (2) to (3) depends upon the area ratios at the two points, and there will be a loss in total pressure head due mainly to losses at the elbow. The drop in pressure and temperature from (3) to (4) depends on the air flow and throttle position.

When the ratio of the absolute pressure at (4) to that at (3) drops to the critical, about 0.53, and the velocity approaches the velocity of sound, the drop in temperature at the center of the air stream, between those points, will be about 95°F. This temperature drop may seem large, but this condition has been established in other equipment involving the same process. The cooling effect on metal parts is only a small fraction of this because without heat transfer the temperature of the boundary layer at the wall surface would be very near the temperature at (3). After (4), the flow area suddenly increases, resulting in a large loss in total pressure head and an increase in temperature. Without fuel flow the temperature at (5) or (6) will nearly equal that at (3) because the difference in velocities is not sufficiently great to cause a large change in kinetic energy.

With fuel flow, the difference in temperature of the air at (5) and the mixture at (6) below the fuel nozzle may be as great as 60°F. due to evaporative cooling if there is sufficient distance between them to establish equilibrium of the evaporating fuel. It is doubtful if the mixture has reached equilibrium at (6), and no accurate method of measuring a specified temperature at that point is available. Operating experience and the results of these and other tests indicate that the usual methods of measurement do not give an accurate indication of temperature conditions of the wet mixture there.

Description of Ice

Ice may be formed in the induction system by three different processes. *Impact Ice* is that formed from water which originally existed in the atmosphere as snow, sleet, or sub-cooled liquid, and includes that formed from liquid water impinging on surfaces that are at temperatures below 32°F. *Throttle Ice* is that which is formed at or near the throttle in a part-closed position due to the cooling effect of the increase in kinetic energy of the air in the restricted flow area. *Fuel Evaporation Ice* or simply *Fuel Ice* is defined as that which is formed due to the cooling effect of the fuel evaporating after it is introduced into the air stream.

Impact ice collects near a surface which changes the direction of the airflow, so that the water or ice particles with a density much greater than the air are carried to the surface with a velocity relative to the air stream. The most dangerous impact ice is that which may collect on the metering elements of the carburetor, affecting fuel-air ratio. Other critical points are the preheater valve and walls of the scoop near it, the roof of the scoop elbow, and screens.

Throttle ice which collects on metal parts is mainly formed from particles which freeze outside of the boundary layer, and are carried to metal surfaces such as the throttle butterfly by their initial momentum. Because of the low capacity of the throttle temperature drop in cooling metal parts, the maximum temperatures at which throttle ice formed was only about 37°F. Ice forms in the central part of the air stream at carburetor air temperatures above 37°F., but metal temperatures were too great for the ice to collect on it.

Fuel evaporation ice probably occurs most frequently in actual operation, because it may form at carburetor air temperatures considerably above 32°F. Most of the heat necessary to evaporate the fuel is supplied from the air as it drops in temperature during the process. Fig. 3 gives the calculated results of limiting initial icing conditions for a specified amount of ice, after the fuel, air and water reach equilibrium at a temperature of 32°F. Fuel evaporation ice may affect air flow by blocking off the blower throat, affect fuel-air ratio by interfering with fuel flow, and affect mixture distribution or quantity of mixture to individual cylinders by upsetting the fuel flow distribution at the fuel nozzle distributor, or air flow distribution in the blower throat.

Calculation of Limiting Icing Conditions

A simple method of calculating the equilibrium state after fuel is evaporated into air with any amount of water was developed using enthalpys of the fuel, air and water. The method of calculation and results will be presented in a future paper, and results of a short series of calculations for the fuel used in these tests are given in Fig. 3. Ice was actually formed with the X-bar fuel distributor at carburetor air temperatures above the calculated limit for a fuel-air ratio of .25. This indicates that the effective fuel-air ratio for icing may be much greater than the average equilibrium value.

Icing Incidents

It is difficult to obtain even an approximate idea of the probable frequency of induction system and carburetor icing for the reason that many of these icing incidents are comparatively mild and either pass unnoticed, or are not reported by the pilots. The more advanced cases of icing, which cause serious malfunctioning of the engine, are comparatively few in number, but they cause a great deal of concern, and should be absolutely eliminated.

The following are histories of several flight incidents which have occurred:

1. Cruising at approximately 10,000 feet with an outside air temperature of 35°F., 57 per cent power, and 80 per cent of rated r.p.m. with full-cold carburetor air, the airplane was drawn into the updraft of a thunderstorm where very heavy wet snow was encountered. On entering the turbulent clouds the pilot throttled his engines to approximately 30 per cent of power, maintaining the original 80 per cent of rated r.p.m. On encountering downdraft and breaking free of the clouds, the pilot opened his throttle, but found his engines were dead. On descending with dead engines from 10,000 feet to about 2500 feet, where the outside air temperature was 65°F., the ice cleared and power was restored.

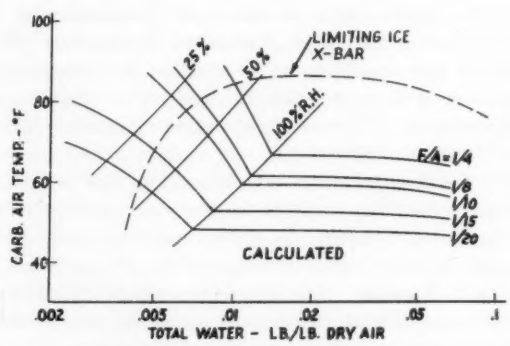

FIG. 3. Calculated adiabatic limiting icing conditions.

2. Cruising at approximately 8000 feet with an outside air temperature of 53°F. to 58°F., 60 per cent power and 80 per cent of rated r.p.m. with full-cold air, manifold pressure dropped 1.0 in. Hg in 20 to 25 seconds. Application of partial preheat to give 75°F. carburetor air temperature removed the ice, restored normal engine operation and prevented ice from forming again. The weather was clear but very hazy and may have had a moisture content greater than a relative humidity of 100 per cent.

3. An airplane was making an instrument landing approach through a winter overcast, at an altitude of

500 feet, an outside air temperature of 38°F. and with 47 per cent power at 83 per cent of rated r.p.m. A loss in manifold pressure of more than 1.0 in. Hg occurred and preheat was applied to give 100°F. carburetor air temperature for one minute. This restored normal engine operation.

4. In another case, only general information was available. All but one engine of the airplane stopped after full preheat of a low capacity was applied in a snowstorm with a low strut temperature. It was fairly well established that the cause of the difficulty was ice below the carburetor. This illustrates the hazard of low capacity preheater systems in which the preheat may be only sufficient to produce a serious fuel evaporation icing condition in a snow or sleet storm.

Comparison of Operating and Laboratory Conditions

Variables of the wind-tunnel study could not be very well controlled, but it is estimated that icing conditions were more severe than those in flight and were sufficiently well controlled to determine the effect of design changes on impact icing of all parts of the scoop.

Known important atmospheric and engine operating variables which affect the formation of ice studied in York were: air temperature, pressure, water content and state, fuel-air ratio, carburetor air flow, engine speed, oil temperatures, fuel temperature. All tests were made with a specially selected clear 73 octane fuel of the same composition. Properties of the fuel, which was stored in sufficient quantity for all tests before they started, are given in Table 9. It is estimated that the control and measurement of these variables and the effect of other ones were such that the laboratory results of limiting icing conditions will agree with those in flight to ±5°F. Obviously, the accuracy of actual temperature measurements was better than this.

Description of Tests

A preliminary program, outlining the plans for this study, was submitted to airlines, airplane and carburetor manufacturers, and the N.A.C.A. Sub-committee on Induction System De-icing before testing was started. The comments received were given consideration, and were extremely valuable in determining the scope and method of test.

The final program outlined three groups of tests to be undertaken: flight tests, wind tunnel tests, and engine and carburetor tests.

Flight Tests

The object of these tests was to obtain induction system metal temperatures during actual flight so that the laboratory results could be correlated with flight operation.

This part of the investigation was made with a Vultee type AB-2 airplane powered by a Pratt & Whitney R-1830-C4 engine, and equipped with a full complement of instruments, including air flow measuring equipment, for accurate engine flight tests. There were only minor design details in the flight test induction system that differed from the system tested in the laboratory. These differences had little effect on the application of the flight test data to the laboratory test procedure.

Sixteen thermocouples were installed in the engine intermediate rear case for measuring the temperature of the walls of the blower throat. A satisfactory method was developed for the installation of metal temperature thermocouples and this same method was used in the laboratory portion of the program.

The effect of four variables on induction system metal temperatures was determined. These were: oil-in temperature, fuel-air ratio, cowl flap angle, and carburetor air temperature. The oil-in temperature was varied from 132°F. to 187°F., the fuel-air ratio from .075 to .090, the cowl flap angle from 6° to 15°, and the carburetor air temperature from 27°F. to 37°F.

The method of test was to stabilize important engine operating conditions and to hold these conditions for at least 15 minutes before data were recorded. The following variables were standardized: altitude, air speed, engine speed, engine power, carburetor mixture setting, carburetor air flow.

Wind Tunnel Tests

The impact icing tests were conducted in the refrigerated wind tunnel at The B. F. Goodrich Company under essentially the same test conditions that have proved successful in the development of de-icer boots.

The object of these tests was to investigate the icing characteristics of an airscoop design, and to determine the design changes that might be necessary to insure satisfactory scoop operation in impact icing conditions.

An assembly view of the apparatus before installation in the tunnel is shown in Fig. 4. Details of the induction system are shown in Figs. 2 and 8.

The induction system, aside from the scoops, consisted of a scoop adapter, a Bendix PD-12F3 carburetor, a Pratt & Whitney R-2000 intermediate rear case and supercharger inlet vane insert. Two variations were made in the original system by replacing the balanced valve with a flapper type valve, and adding a two-inch plenum chamber on the rear of the scoop. The redesigned scoop utilized a combination rotary and flapper type valve and a plenum chamber.

The tunnel water spray system consisted of a spray bar, electrically driven water pump, compressed air supply, water reservoir and platform scale. The water spray jets were interchangeable and were spaced eight inches apart along the bar. Three sizes were used. The platform scale was used to determine the total water input into the tunnel during a test. This system was standard equipment of the tunnel.

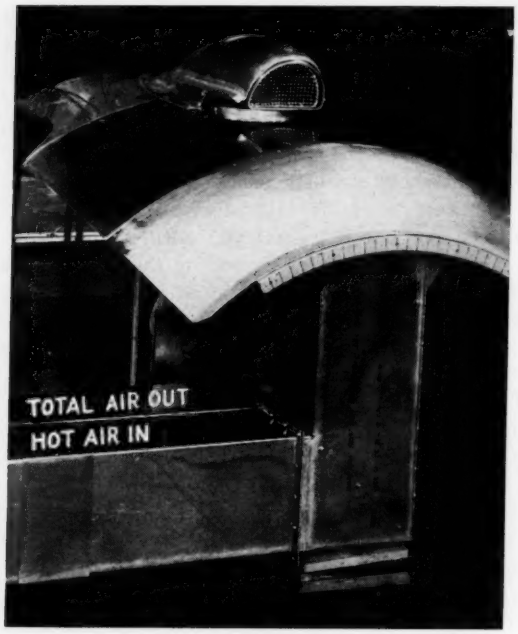

FIG. 4. Assembly view of apparatus before installation in the wind tunnel.

The exhaust system for carburetor air and the hot air supply system were designed and operated to obtain carburetor air flows up to and including the flow required at cruising speed. No attempt was made to obtain larger air flows because of the low tunnel speed of 80 m.p.h. Hot air temperatures well over 100°F. could be attained by the two steam heaters. The duct arrangement was such that the exhausted air could be heated and returned to the system as preheat, or could be returned directly to the tunnel.

An apparatus for approximately determining the free moisture concentration was developed during the testing. Tunnel air was drawn through a heated sampling tube and separation canister with tunnel velocity at the entrance. The canister contained glass wool for separating the water from the air, and was weighed before and after the run to determine the weight of water collected.

Before the icing runs were made, a survey was made of the tunnel velocity distribution at the throat outlet and in a plane at the scoop entrance. Contours of velocity distribution were plotted. Airflow measurement using A.S.M.E. flat plate orifices was checked by pitot tube transverses to ±3 per cent.

Tunnel Test Procedure

The general procedure for making an icing test was as follows: (a) Install special equipment and insert thermocouple probes in proper locations. (b) Set heater valve position and adjust heaters to give degree of preheat desired. (c) Adjust dampers in air flow system. (d) Install proper jets in spray bar and determine initial weight of spray water. (e) Set up photographic equipment. (f) Disengage tunnel air circulator and start blower and tunnel propeller motor. Set tunnel air speed at maximum. (g) Adjust blower speed to obtain carburetor air flow desired, insert spray bar, and start water spray system pump. (h) Make photographic record of tunnel conditions and entrance ice cap growth during test. Make periodic readings of system air flow and metal temperatures, and prints of static pressure levels. (i) Stop blower, tunnel propeller and spray system when tunnel temperature gets too high to form ice. Obtain final weight of spray water. (j) Photograph ice formation from front, remove section of scoop and inspect and photograph the inside of the system. (k) Remove all ice formations and start tunnel air circulator in preparation for next run.

The icing characteristics of the original scoop and balanced valve were determined under the following variable conditions: (a) jet size: large, medium, small; (b) carburetor air flow: minimum 2760 lbs. per hr., maximum 7200 lbs. per hr.; (c) valve position: cold, 1/3 open, full open with 100°F. hot air.

After the completion of the above tests, it was decided that large jets and cruising air flow represented the most critical icing conditions so that the majority of the subsequent tests were made that way. The original scoop with a flapper type valve was tested with the same valve positions outlined above. In addition, preheats at 50°F. and 100°F. were used. In some of the runs on these scoops, the ice caps at the scoop entrance were removed after they had become 2 or 3 in. thick. This was justified by the fact that on an actual installation, vibration would probably shake the caps off. By removing the caps, ice that would have been deflected by them was allowed to enter the scoop, thus increasing the severity of the test.

As soon as enough data had been collected on these systems to determine their most serious icing faults, a conference was held to discuss scoop and preheat requirements. This conference was attended by all interested parties and recommendations were made for the redesign of the scoop. The redesigned scoop was fabricated according to these recommendations and tested under the same conditions used for the tests of the original scoop.

The icing of the entrance screen was determined with three different jet sizes and performance of the carburetor screen was observed in all runs after Run 10.

A streamlined shield was tested with the following variations of conditions: (a) jet size: large, medium, small; (b) carburetor air flow: minimum 3400 lbs. per hr., maximum 6400 lbs. per hr.; (c) location of trailing edge ahead of scoop: minimum 1.5 in., maximum 5.5 in. from the scoop entrance.

When the outside air temperature became too high for the refrigeration system to maintain icing conditions, dry ice was put in the tunnel. This was done for most runs after Run 12.

Effect of Redesigned Scoop on Carburetor Metering

At the conclusion of the wind tunnel tests, the effect of the scoop on carburetor metering was determined by air flow tests. In these tests there were pulsations in carburetor metering pressure and air flow attributed to the plenum chamber. The pulsations were eliminated on the test stand by blocking off the plenum chamber with a baffle, but it was not apparent that they were caused by the plenum chamber alone. Because the plenum chamber improves velocity and temperature distribution of cold or partially preheated air in the carburetor and reduces undesirable ice and rain, it was decided that flight tests should also be made on it.

Although there was no serious difficulty with engine operation in flight, the effect of pulsations was noted there also. It is believed that these could be eliminated by further development, and because of the excellent qualities of the plenum chamber it may be desirable to attempt to eliminate the pulsations by improving the internal aerodynamic design of the scoop.

The redesigned scoop with and without the plenum chamber was satisfactory from a carburetor metering point of view, but the present design of plenum chamber is not recommended because of the fluctuations which were produced in both carburetor metering pressure and intake manifold pressure.

York Tests

The object of these tests was to determine the limiting icing conditions for all important operating variables with two types of fuel nozzles, to determine the preheat requirements necessary to eliminate all icing hazards in the engine section of the induction system, and to investigate the effectiveness of other de-icing agents relative to preheat.

The equipment is illustrated in Figs. 5, 6 and 7. The refrigeration equipment was originally designed for cold weather operation and consisted of three compressors and three stages of cooling. For operation in summer weather, an additional cooling stage and compressor was added, providing a capacity that would cool carburetor air flow of 6000 lbs. per hr. from approximately 60°F. to −30°F.

A complete R-2000 rear section was mounted in a specially constructed fire-proof test room and was driven through a speed increaser by a 150 hp. variable speed a.c. motor. Complete instrumentation was provided in order that all important variables could be accurately controlled and measured.

The carburetor air supply system consisted of two 8 in. ducts. The cold air duct contained steam and water jets for obtaining the desired water concentration, gate valves for altitude throttling, a steam coil for fine temperature adjustment, and orifices for measuring air flow. The hot air duct was similarly equipped, and hot air temperatures well over 100°F. could be obtained from the steam heating coils.

Facilities were provided to obtain a complete control over the flow, temperature and pressure of the fuel. The fuel was supplied to the carburetor in such a manner that it could be by-passed when not required, or it could be readily introduced into the air through either of the two discharge nozzles without removing the carburetor.

Approximately 250 pressure and temperature measurement points were installed throughout the system. Pressure lines were made of copper and neoprene tubing and were connected to U-tube manometers, slant gages or dial gages. Temperature measurements were made with iron and constantan thermocouples connected to a ten point recorder, an indicating recorder or a portable precision potentiometer.

An alcohol supply system was provided which would deliver alcohol to the induction passage at any desired flow rate. This system consisted of a reservoir and platform scale, electrically driven pump, flow measurement equipment and four swirl type mist nozzles. These nozzles were located at the inner radius of the scoop elbow and the rear wall of the scoop as shown in Fig. 7f. Each of these nozzles could be shut off independently of the other and several combinations of jets could be used to determine the best for removing ice.

The induction system used was similar to that used for the first impact icing tests at Akron, except that a complete Bendix-Stromberg PD-12F3 pressure type carburetor and a complete engine rear section were operated. The carburetor was mounted with flexible external connections so that it could be removed readily. A dismantling procedure was developed so that the blower throat could be inspected from 30 to 60 seconds after the supercharger driving motor switch had been opened.

An elaborate calibration program was completed to obtain the most effective method of instrumentation, to determine the overall equipment performance, and to check all metering devices.

The blower throat metal temperatures obtained in the flight tests were checked with those obtained in the laboratory under similar conditions and were found to agree satisfactorily.

In general the procedure for obtaining the conditions required for the limiting icing condition test program was as follows: The desired carburetor air flow rate was adjusted by throttling the air flow ahead of the scoop and the desired water concentration obtained by calculating and injecting the proper amount of steam and water. Throttle position, oil-in temperature, carburetor mixture setting, and scoop pressure were adjusted to the desired condition. The fuel flow was regulated to the proper amount and by-passed back into the storage tank with no fuel flow into the carburetor until the entire system had reached equilibrium at the desired temperature for that particular run. When the equilibrium conditions had been reached the fuel was introduced into the induction system through

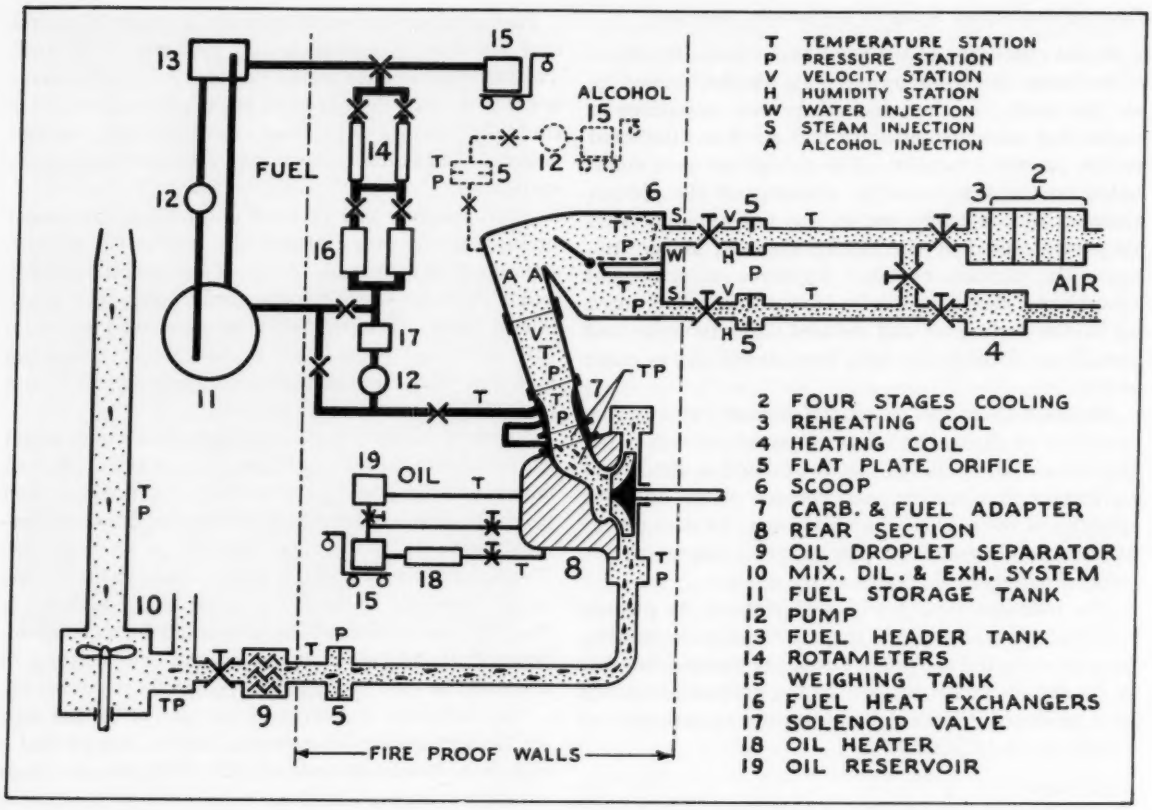

FIG. 5. Schematic layout of York laboratory apparatus.

either type of fuel discharge nozzle. The instrument readings which were pertinent to the particular test were recorded at set intervals. If necessary, minor adjustments were made throughout the run to maintain the desired test conditions.

After the test had continued for a length of time believed to be adequate for consistent and comparative results, the equipment was shut down and the induction system dismantled and examined for ice formations.

The prevention of ice with heat was studied with the following variable conditions: five average carburetor air temperatures, two water concentrations, two total carburetor air flows, two partial heater valve positions and two throttle positions.

The temperature difference between hot and cold air was held to 50°F. ± 2°F. The static pressure difference between hot and cold air was held at eight inches of water.

The determination of the preheat de-icing performance was done by allowing ice to form in the system until an appreciable drop in air flow was obtained and then opening the heater valve to full hot and recording the time required to obtain full throttle air flow. This procedure was followed using five different hot air temperatures and three different types of ice.

Removal of ice with alcohol was studied by forming ice in five different ways and noting the time required to restore full throttle air flow after injecting alcohol at six different flow rates. Three separate alcohol injection systems were used and their relative de-icing performance was checked.

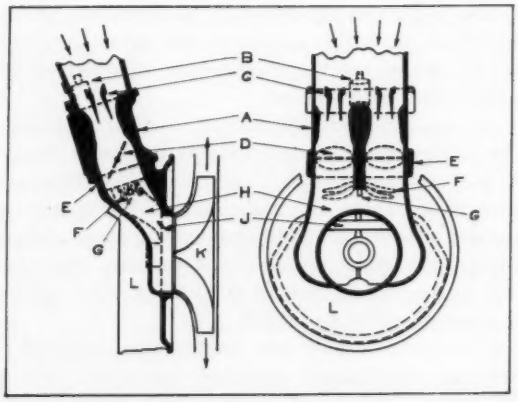

FIG. 6. Carburetor and engine induction system. *A*, throttle body; *B*, automatic temperature compensating unit; *C*, boost venturi; *D*, throttle butterfly; *E*, fuel adjustment; *F*, X-bar fuel distributors; *G*, X-bar fuel discharge nozzle; *H*, blower throat; *I*, impeller inlet vane; *K*, supercharger impeller; *L*, supercharger intermediate case.

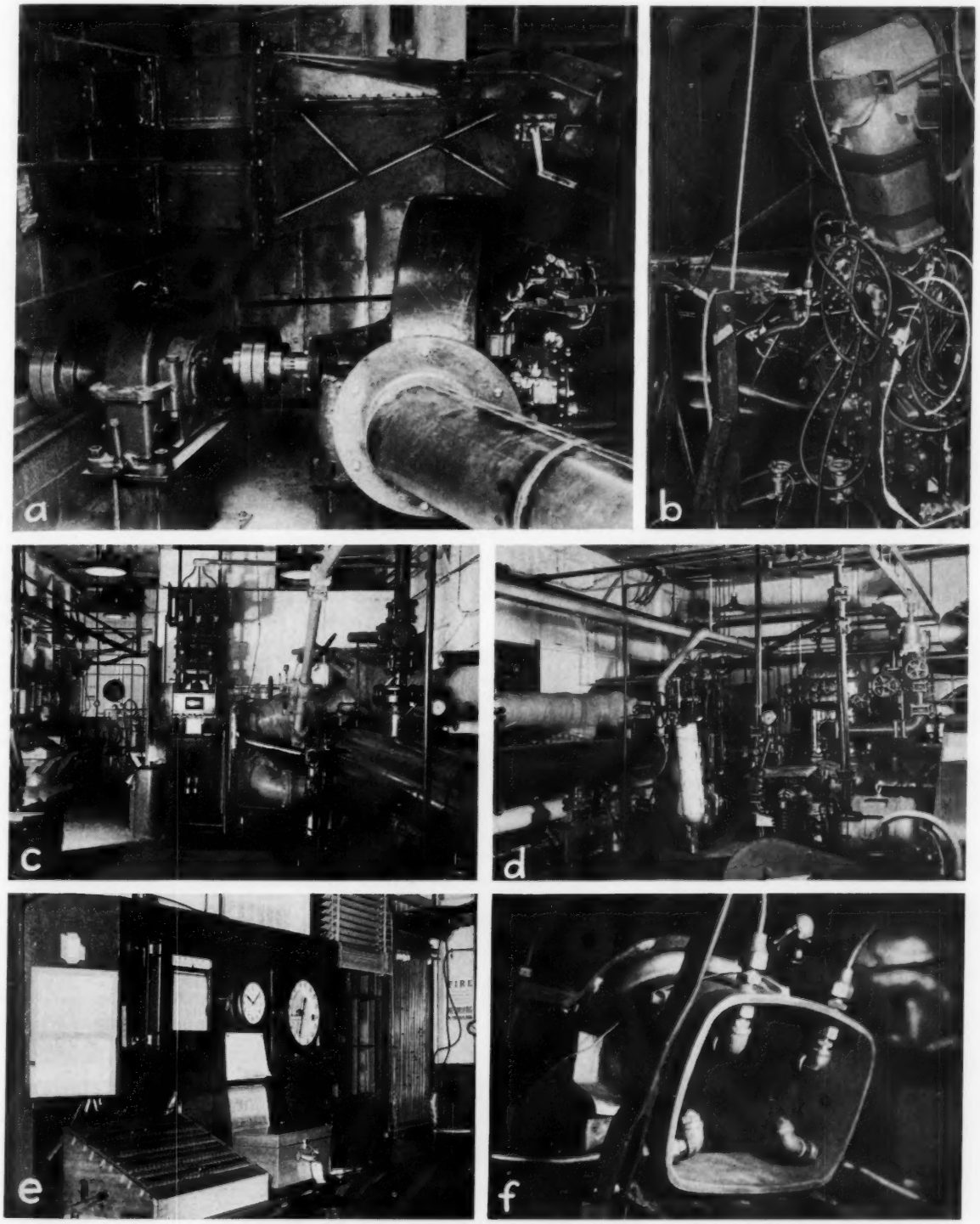

FIG. 7. York equipment. *a*, general view of engine rear section before instrumentation; *b*, same (*a*) after instrumentation; *c*, general view of laboratory looking toward engine room and control station; *d*, view of laboratory showing refrigerating coils, compressors, etc.; *e*, temperature measurement station showing thermocouple selector switch, Ten Point Micromax Recorder, Micromax Indicator Recorder, and Portable Precision Potentiometer; *f*, view up into scoop elbow showing location of alcohol injection nozzle.

Impact ice was formed by maintaining a carburetor air temperature well below 32°F. and by injecting steam into the cold air duct ahead of the scoop, producing large quantities of snow. Throttle ice was formed with carburetor air temperatures from 32°F. to 37°F., throttle angles below 45°, and carburetor air moisture concentrations in excess of 100 per cent relative humidity. This ice formed on and in the neighborhood of the throttle butterflies. Fuel evaporation ice was formed with the X-bar fuel nozzle with carburetor air temperatures below 90°F. and with the water concentration of this air in excess of 100 per cent relative humidity.

A log of observed and photographed results was kept throughout the tests.

Analysis of Results

Limitations of Wind Tunnel Tests

In appraising the results of the refrigerated wind tunnel tests, the following limitations should be considered:

(a) The tunnel velocity was 80 m.p.h.

(b) Only a very few types of ice could be formed at a relatively high temperature, so that the prediction of the effect of fine snow and sub-cooled liquid water could not be determined.

(c) The total concentration of the liquid plus solid water in the tunnel was approximately .008 lb. per lb. of dry air for large jets, and .0043 lb. per lb. of dry air for small jets, as determined by sampling equipment. These results are open to question because there was not sufficient time to perfect this type of equipment.

(d) The system was not subjected to the vibration that would be encountered in an actual installation.

(e) The pressure differential between the hot and cold air was less than that under actual flight conditions.

(f) The propeller slip stream effects were not duplicated.

(g) Constant tunnel temperatures could not be maintained.

(h) Heat transfer from parts adjacent to the scoop was not duplicated.

Icing of the Original Design of Scoop

Runs 1 through 7 of Table 3 established the icing characteristics of the original scoop with the heater valve in full cold position. The general character of impact ice formation was also determined by these tests.

The first series of runs established the conclusion that the balanced valve should not be used in icing conditions. The balance and the struts supporting it iced rapidly and the formations on these parts prevented operation of the valve after only a few minutes of icing. These formations also deflected ice so that it collected on the scoop walls and roof where it was a further obstruction to valve operation. Fig. 8e is an example of this ice formation.

Runs 9 and 10 were made to determine icing characteristics with the balanced valve in partial and full preheat positions. These runs indicated that there was not enough heat transferred to the balance to prevent its freezing to the scoop walls because of the ice that formed on it. As in the full cold position, the valve became inoperable after only a few minutes of icing. It was also observed that this valve caused severe stratification of the hot and cold air. Ice formed on the walls and back of the scoop even when air temperatures of more than 100°F. were measured at the inside radius of the elbow. This condition would probably have considerable effect on carburetor metering. The stratification might also carry through the impeller and cause variations in intake temperatures and distribution.

It was also noted that the scoop entrance area could be reduced by about two-thirds before there was an appreciable drop in air flow. This is an indication that the scoop was adequately designed for altitude and little or no ram flight conditions.

Variations of air flow from 2760 lbs. per hr. to 7200 lbs. per hr., with large, medium and small jets, indicated that the effect of air flow was not very marked but there appeared to be a slight tendency for ice to form more rapidly at the entrance with low air flows and to collect in greater quantities beyond the scoop elbow with high air flows. Ice formed from small jets deposited further down into the system and larger jets iced the entrance more rapidly because of higher water input rate. There was no distinct difference in the type of ice formed using various jet sizes.

The temperature range of the tunnel air throughout a run obviously had a greater effect than the jet size on the type of ice formed. Since the rate of change of tunnel temperature could not be controlled, scoop and valve performance could not be studied on the basis of the effect of the various kinds of ice.

On several occasions, after the ice formations were photographed and examined, enough ice was removed to allow the valve to be moved to full hot. The system then was de-iced with hot air and it was noted that large chunks of ice were dislodged, causing sharp fluctuations in air flow. It is possible that without a carburetor screen, these chunks could seriously damage the venturis and impact tubes in the throttle body.

Icing of the Entrance Screen

Preliminary tests indicated that screen icing was most rapid with medium jets. In Run 8, with medium sized jets, the screen iced in 1 minute and 40 seconds. The effect of entrance screen icing is shown in Fig. 8f.

Performance of the Flapper Valve

Runs 11 through 17 determined the performance of the flapper type valve. In evaluating the results of these tests it should be considered that the valve was rather hurriedly made and embodied design characteristics, the correction of which would improve its

FIG. 8. Scoops and scoop icing tests. *a*, original scoop design without value; *b*, original scoop design with plenum chamber added; *c*, redesigned scoop, heater valve open; *d*, redesigned scoop, heater valve closed; *e*, icing of balanced valve in cold position; *f*, icing of entrance screen; *g*, icing of balanced valve in partial preheat position; *h*, ice bridge ahead of flapper valve; *i*, ice formation on roof of scoop; *j*, streamlined shield; *k*, effect of shield; *l*, ice cap on redesigned scoop; *m*, internal icing of redesigned scoop.

performances greatly. Also some of the most serious difficulties encountered with this valve were caused by faults in the scoop design.

With the valve in full cold position, ice formations on the scoop walls and roof prevented it from being moved to the full hot position after 10 to 20 minutes of severe icing. One of these formations is illustrated in Fig. 8i. The situation is not too serious because it is unlikely that a pilot would fly in an icing condition for that length of time before using preheat. However, as many critical conditions as possible should be eliminated so that this situation was regarded as intolerable. Probably it could be eliminated by proper design of the walls and roof of the scoop, so that ice would not form in the region of the valve operation.

With intermediate preheat valve positions, a considerable amount of ice formed ahead of the valve, as shown in Fig. 8h. The heat transfer through the valve was enough to free it from this ice after a sufficient amount had formed to insulate the valve from the cold air blast, but the ice formation remained intact preventing movement to full hot, restricting cold air flow considerably. Higher valve temperatures would have prevented the formation of most of this ice, eliminating the hazard of faulty valve operation and the danger of large chunks of ice sloughing into the adapter. These temperatures could be obtained by extending the heat transfer area of the hot side and by increasing the heat transfer factor of the hot air. When 50°F. instead of 100°F. preheat was used, the amount of ice formed ahead of the valve increased. The preheat temperature required is dependent on the heat transfer capacity of the particular valve design.

Stratification of hot and cold air was also severe with the flapper valve. This condition is perhaps the most difficult one to correct in flapper valve design. The best would be one that places the valve butterfly in a nearly vertical position when partial or full preheat is desired.

During Run 17 the valve hinge froze making operation impossible when the valve itself was free. This indicates the necessity of locating the hinge or hinges where they would be protected from ice. Rain should also be kept off the hinges since there is a possibility of going from rain into a freezing condition and a wet hinge might become inoperable. This valve was not developed further because others seemed to show more promise.

Performance of the Plenum Chamber

For Runs 18 through 21, the original scoop with a two inch plenum chamber was installed (Fig. 8b). Ice formed on the sloping roof ahead of the chamber, thus preventing most of the ice from collecting in it. On de-icing the system with preheat after a full cold run, it was noted that large chunks of ice, that previously had sloughed into the adapter, were caught in the plenum chamber. During Run 18, about an inch of fine snow collected in it.

In a special rain collection test, this scoop collected over 300 per cent more water than the original one. With cruising air flows the velocity and temperature distribution was improved at the carburetor inlet regardless of the heater valve position.

The improvements observed in velocity distribution, temperature distribution, and ice and rain collection led to the incorporation of the plenum chamber in the redesigned scoop. Temperature and velocity distributions at the carburetor inlet, using the original and redesigned scoops, are compared in Figs. 9 and 10.

Performance of the Icing Shield

Performance of a streamlined shield was studied in Runs 22 through 25. By examining the formations on the entrance screen, which was used as an icing indicator, the amount and the angle of incidence of the ice entering the scoop was obtained. An optimum location was found for the shield, and at this position insufficient ice formed on the screen after 15 minutes of icing to affect the air flow appreciably. The amount of water collected in a rain test was negligible compared to that collected without the shield.

Icing of the Redesigned Scoop

The redesigned scoop was thoroughly tested under the most severe conditions attainable in Runs 26 through 36. No operating difficulties were encountered in any of the runs.

The thinner leading edge on the redesigned scoop entrance accelerated ice formation at this point. The rapidly formed ice caps shielded the inside of the scoop, reducing the amount of ice collected there. The bond between the scoop and the ice caps was weaker and it was assumed likely that even though the caps formed faster they would not remain attached to the scoop as long in actual operation, and the shielding effect would be a minimum. From an icing standpoint it would be more desirable to have a thick leading edge on the scoop entrance since it would reduce the hazard of caps falling into the scoop; this design would also be better from an aerodynamic point of view.

The redesigned scoop had a higher aspect ratio, which reduced the amount of ice that entered the system, because the entrance was beneath most of the wake coming off the front of the cowl.

No undesirable ice formations were noted inside the scoop on full cold runs, except for Run 31. At this time a large chunk deposited in the region of valve travel preventing operation of the valve. In Run 33 it was established that this formation was due to the method of removing the ice caps. During Run 31 the caps were knocked off with a rod and part of a cap evidently was pushed into the scoop. When the caps were picked off by hand, this condition did not recur.

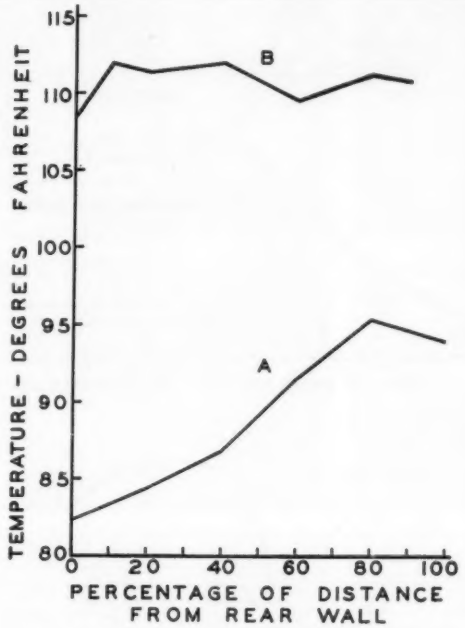

FIG. 9. Comparison of temperature distribution in scoop adapter. *A*, original scoop, flapper valve two-thirds open; *B*, redesigned scoop, valve two-thirds open.

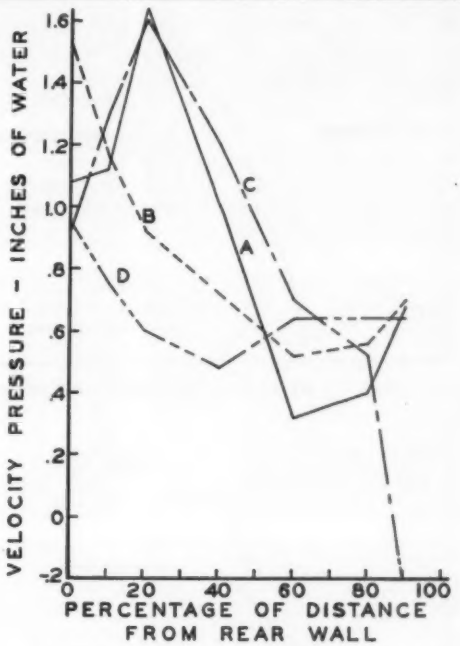

FIG. 10. Comparison of velocity distribution in scoop adapter. *A*, original scoop, flapper valve closed; *B*, redesigned scoop, valve closed; *C*, original scoop, flapper valve two-thirds open; *D*, redesigned scoop, valve two-thirds open.

During full hot operation (Runs 34 and 35) no ice formed on the cold air shut-off valve and very little ice formed ahead of it. The ice caps were removed from the front of the scoop to increase the severity of the test.

Analysis of Engine and Carburetor Icing Tests

The difference between blower throat metal temperature readings in the laboratory and in flight was so small that it may be assumed that this variable was satisfactorily simulated.

Particle size of liquid water injected into the carburetor air stream was estimated to be about 0.2 mm. by a very rough extrapolation of the Johnstone and Brown data on the performance of mist nozzles.

An analysis was made of the per cent of fuel and water in ice samples with an X-bar fuel distributor, and carburetor air temperatures of 50°F. and 32°F. and mixture settings of Auto-Lean and Auto-Rich. The percentage of fuel was greater at the 50°F. carburetor air temperature and with the Auto-Rich mixture settings. Fig. 13o illustrates that ice formed in the blower throat contains sufficient fuel to burn. Note the amount of ice which had formed.

Using an X-bar fuel distributor, fuel evaporation ice will form at carburetor air temperatures up to 80°F. when the water concentration varies over a considerable range, and under a wide range of engine operating conditions. Under the most severe conditions, ice will build down from the X-bars and nozzle, and build up from the vanes until the two formations meet and close off the blower throat. This type of ice is illustrated in Figs. 12a and 12b.

With the X-bar fuel distributor the effect of fuel-air ratio, fuel temperature, scoop pressure and oil-in temperature is relatively slight, but somewhat greater effect was found for various air flows and throttle positions over a wide range of air temperatures, but the effect of all those was very small compared to that of carburetor air temperature and water concentration. These results are illustrated in Figs. 14 through 20.

By observing the location of ice formation and the variations in temperature of the blower throat walls, with the X-bar distributor, it does not seem practical to eliminate this ice by heating critical metal parts.

Using the new fuel nozzle no conditions were found which would produce sufficient fuel evaporation ice to affect air flow appreciably, although engine distribution might be affected slightly. With this type of nozzle no dangerous fuel evaporation ice was formed but dangerous throttle ice and impact ice did form in the carburetor.

Throttle ice was formed at throttle openings of less than 45°F. and with carburetor air temperatures between 32°F. and 37°F. This ice may form very rapidly causing a sharp drop in carburetor air flow and it is possible that the throttle would become inoperable. This type of ice is illustrated in Fig. 13. It is difficult to see on the photograph because it is hard and clear. Table 2 indicates the large effect of throttle pressure drop, or throttle angle on the formation of throttle ice.

Impact ice, consisting of fine snow and sleet, is shown

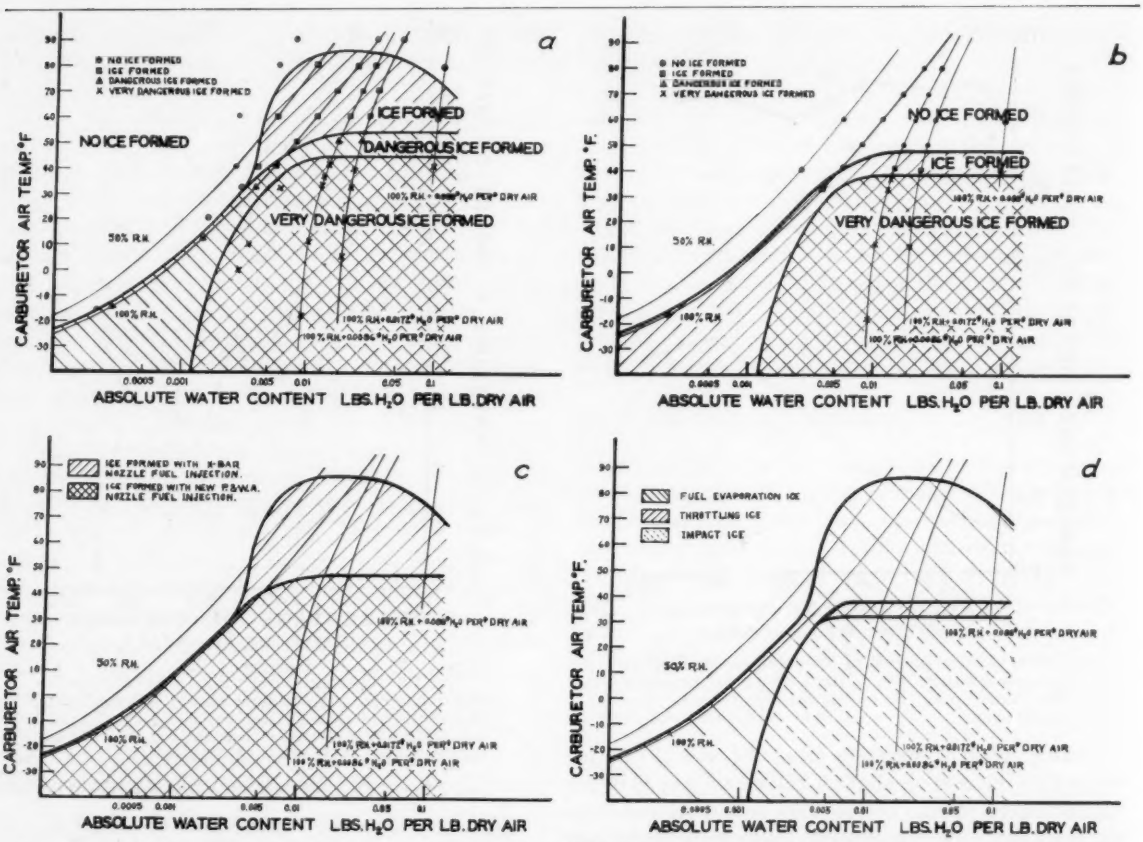

FIG. 11. Summary limiting icing conditions. *a*, limiting icing conditions with X-bar fuel discharge nozzle; *b*, limiting icing conditions with new design fuel discharge nozzle; *c*, comparison of ice formed *vs.* no ice formed conditions using X-bar fuel discharge nozzle and new design fuel discharge nozzle; *d*, regions of formations of fuel evaporation ice, throttling ice and impact ice, using X-bar fuel discharge nozzle.

in Figs. 12 and 13, and was formed with carburetor air temperatures below 32°F. with free water. This type of ice had the most detrimental effect on carburetor performance. This condition requires the elimination of complex obstructions to the air flow and shielding of the metering elements.

It was established in tests of the prevention of ice with preheat that to eliminate all ice from the blower throat with the X-bar fuel nozzle, the preheat system should be capable of supplying 110°F. hot air with a relative humidity of 100 per cent or less at 0°F. strut temperature at cruising power. Using the new fuel nozzle the preheat air requirements can be reduced to 50°F.

TABLE 2

Effect of Throttle Pressure Drop on Throttle Ice Formation

	Throttle Pressure Drop, In.-Hg		Throttle Angle, Deg. 0° = Closed, 75° = Open	
	4000 lb./hr.	6000 lb./hr.	4000 lb./hr.	6000 lb./hr.
Amount of Ice	Air Flow	Air Flow	Air Flow	Air Flow
None	1.65	2.8	46	50
Trace	2.2	3.2	41	45
Slight airflow drop	3.35	4.05	36	40
Definite airflow drop	4.9	6.6	31	35

The tests for the removal of ice indicated that ice formations which produced as much as a 60 per cent drop in carburetor air flow could be removed with a 60°F. hot air in approximately 20 to 30 seconds. With this same per cent reduction in air flow and hot air temperatures over 100°F., the time to restore full throttle air flow becomes almost instantaneous. Two examples of preheat de-icing are illustrated in Figs. 21 and 22.

Using alcohol as a de-icing agent and alcohol flow rates as high as 80 lbs. per hr., the time required to restore full throttle air flow was found to be approximately 100 seconds. Run 192, Fig. 23 and Run 194, Fig. 24 indicate that it is possible that the time to restore full throttle air flow may not be appreciably reduced by increasing alcohol flow above 60 lbs. per hr. The effect of ice heard passing through the impeller indicated that the alcohol dislodged large chunks of ice and that the de-icing time was limited by the ability of the impeller to dispose of it.

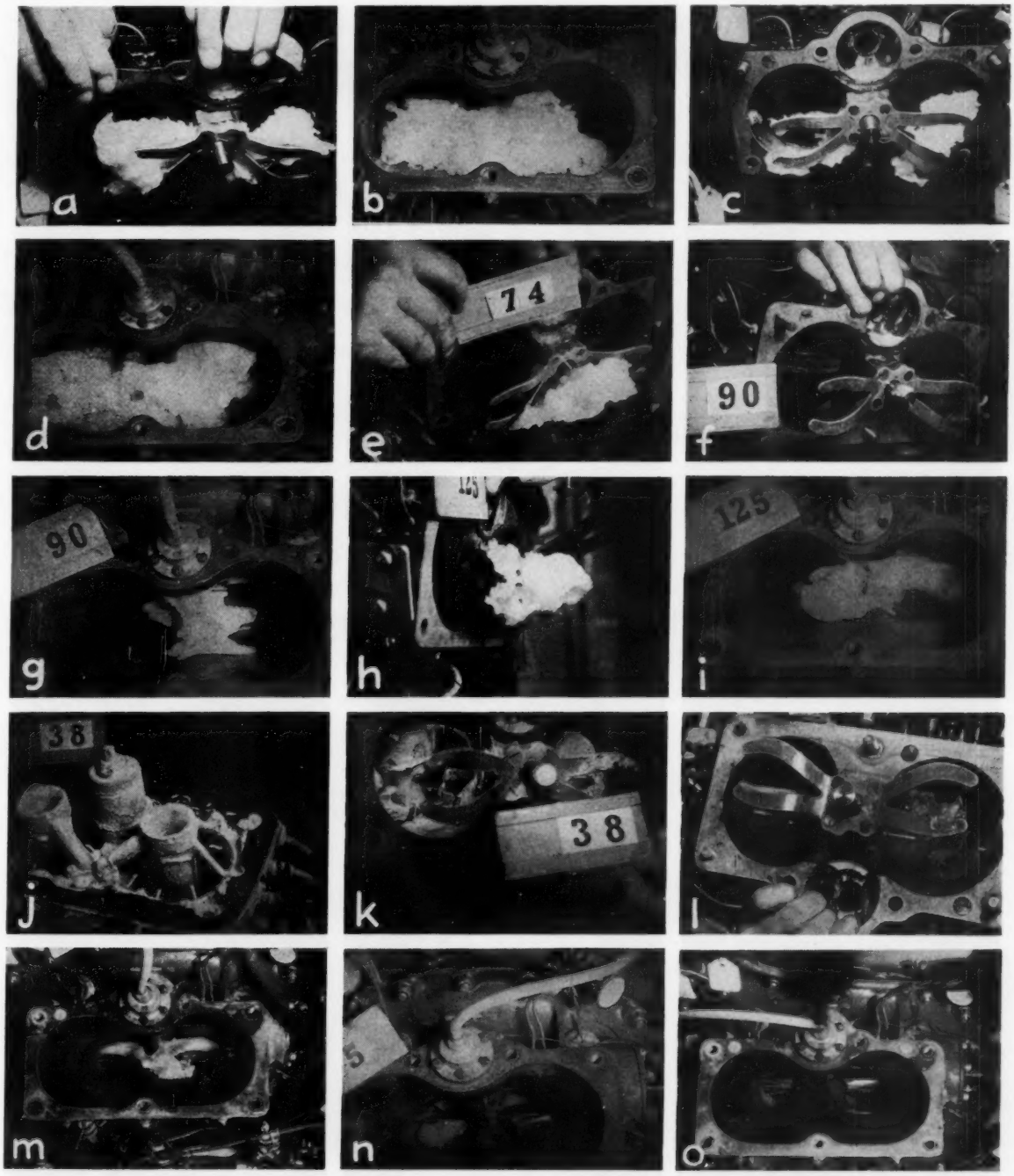

FIG. 12. Carburetor and engine icing. *a* and *b*, Run 58, fuel evaporation ice with X-bar discharge nozzle; *c* and *d*, Run 60, fuel evaporation ice with X-bar discharge nozzle; *e*, Run 74, fuel evaporation ice with X-bar discharge nozzle; *f* and *g*, Run 90, fuel evaporation ice with X-bar discharge nozzle; *h* and *i*, Run 125, fuel evaporation ice with X-bar discharge nozzle; *j* and *k*, Run 38, impact icing with snow; *l* and *m*, Run 239, fuel evaporation ice with X-bar discharge nozzle; *n*, Run 55, fuel evaporation ice forming on obstruction, using new type discharge nozzle; *o*, Run 237, no ice formed, using new type discharge nozzle.

For all conditions in the system when the specified preheat is available and the air flow is greater than 1000 lbs. per hr., preheat is a more rapid de-icing agent than well distributed alcohol flowing at a high rate. Fig. 25 illustrates the alcohol distribution as indicated by the temperature of the mixture.

SUMMARY OF RESULTS

The basis for the Summary of Results are Tables 3 through 8 and Figs. 2 and 8 through 25. It should be understood that results mentioned are for only the particular system tested. Table 9 gives the properties of the fuel used in these tests.

FIG. 13. Carburetor and engine icing. *a*, Run 49, pronounced throttling ice; *b* and *c*, Run 93, throttling ice; *d*, *e* and *f*, Run 37, impact icing with snow; *g*, Run 37, fuel evaporation ice with new type discharge nozzle; *h* and *i*, Run 79, impact icing with snow; *j*, Run 79, trace of fuel evaporation ice with new type discharge nozzle; *k* and *l*, Run 189, impact icing conditions for ice removal runs; *m*, Run 190, throttle icing conditions for ice removal runs; *n*, Run 191, fuel evaporation icing conditions for ice removal run; *o*, fuel evaporation ice burning.

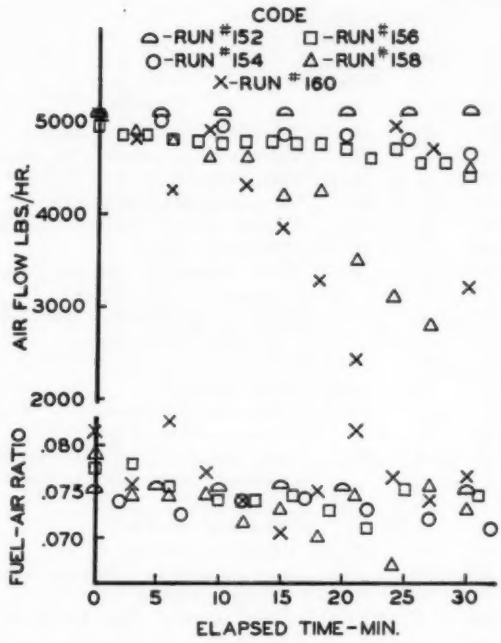

FIG. 14. Effect of water concentration on rate of ice formation with X-bar discharge nozzle.

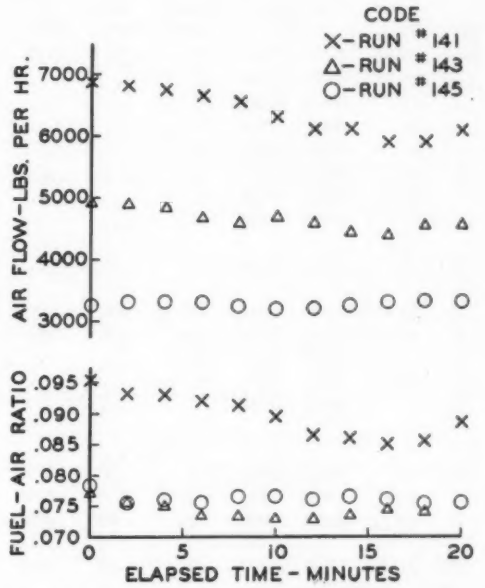

FIG. 15. Effect of airflow on rate of ice formation with X-bar fuel distributor.

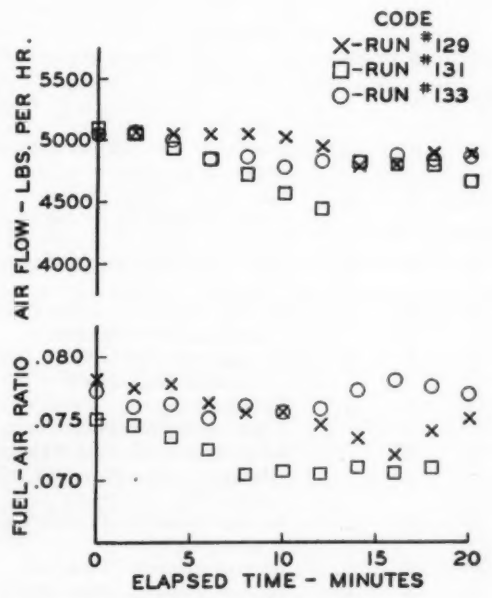

FIG. 16. Effect of throttle position on rate of ice formation with X-bar fuel distributor.

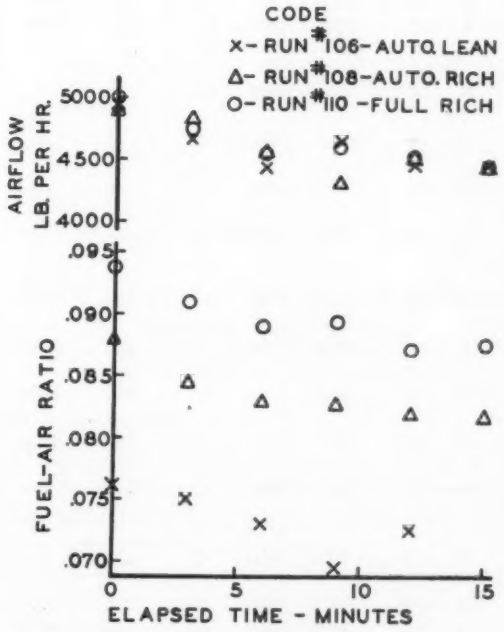

FIG. 17. Effect of fuel-air ratio on rate of ice formation with X-bar fuel distributor.

TABLE 3
Summary of Wind Tunnel Tests

Run No.	System	Time Run, min.	Total Carb. Air Flow, lb./hr. Start	End	Heater Valve Pos.	Hot Air Temp., °F.	Hot Air Flow, lb./hr.	Tunnel Temp., °F. Start	End	Type of Spray	Weight of Jet Water, lb.	Remarks
1	...	32	5000	4500	C	...	..	2	25	2	18	Valve frozen in full cold position
2	...	23	5000	3750	C	...	..	19	27	3	22	Valve could be moved to 1/3 open
3	...	31	5000	5000	C	...	..	10	26	1	13	Valve inoperable
4	...	31	2700	2000	C	...	..	− 2	21	2	23	Valve inoperable after 3 min.
5	1	66	6000	3000	C	...	..	− 7	14	5	44	Valve inoperable. Entrance blocked
6	...	31	6000	3250	C	...	..	− 9	18	6	19	Valve inoperable
7	...	27	6000	5050	C	...	..	−11	20	4	6	Valve inoperable. Impact tubes plugged
8	2	9	6000	3000	C	...	..	0	17	5	..	Icing entrance screen. Min. flow attained, 1 min. 40 sec.
9	1	19	6650	6150	2/3	100	3000	−10	24	5	9	Valve inoperable
10	1	16 1/2	7200	6500	1/3	100	2900	− 2	31	5	6 1/2	Valve inoperable
11	3	18	7300	6400	C	...	..	5	20	5	18	Valve operation difficult
12	3	30 1/2	7300	5800	C	...	..	7*	25	6	36	Valve opened to 3/4
13	3	16	6650	6000	C	...	..	2	23	6	21	Valve opened to 3/4. 100° preheat applied. Valve could be opened in 12 min.
14	3	31	6000	6000	O	100	5700	− 4*	14	5	22	Valve fully operable
15	3	33 1/2	6300	6300	O	100	5700	− 2*	17	6	33	Valve fully operable
16	3	15 1/2	6000	6000	2/3	50	4800	− 2*	15	6	20	Ice bridged valve. Valve closed but bridge remained
17	3	31 1/2	6000	6000	O	50	5300	− 4*	18	6	41	Valve operable but bridge remained
18	4	15 1/2	6000	5750	C	...	..	−10*	15	6	22	Inch of fine snow in plenum. Formation on roof prevented valve opening more than 1/3
19	4	20	6300	5200	1/3	100	2500	−10*	20	6	31	Ice caps removed. Valve open only 2/3
20	4	18	6000	5700	2/3	100	4000	− 4*	20	6	33	Valve frozen. Broken when forced
21	4	20	5250	5250	O	100	4700	0*	21	6	30	Ice caps removed. Valve free
22	5	15	6200	5750	C	...	..	− 4	18	6	20	X = 1 1/2". † Screen not iced appreciably
23	5	15	6100	5600	C	...	..	−14	15	6	20	X = 3 1/2". Screen not iced appreciably
24	5	15	6400	4900	C	...	..	1	25	5	9	X = 5 1/2". About 2/3 of screen plugged
25	5	15	3500	3200	C	...	..	−11	20	6	21	X = 3 1/2". Carb. air flow had little effect on icing of screen
26	6	15	5800	5100	C	...	..	−11*	14	6	24	Valve free. Inside of scoop partially shielded by ice cap at entrance
27	6	15	5400	5400	O	100	4700	− 2*	19	6	23	Valve free. No ice on cold air cutoff
28	6	10	6000	6700	2/3	100	4875	− 2*	23	6	16	Ice caps removed. Valve closed, but would not fully open
29	6	10	5400	5400	2/3	100	4700	− 3*	22	6	16	Ice caps removed. Valve broke formation when opened
30	6	38 1/2	5400	3500	2/3	100	4200	−16	18	6	60	Ice caps removed. Valve would close during run, but not open
31	6	15	6100	5400	C	...	..	− 8*	16	6	..	Ice caps removed. Part of one cap lodged in scoop, prevented valve operation
32	6	7	5400	5000	C	...	..	8*	20	6	..	Ice caps removed. Preheat would not loosen valve
33	6	15	6100	5400	C	...	..	5*	20	6	..	Ice caps removed. Valve free
34	6	15	5800	5400	2/3	50	4650	−10*	14	6	16	Ice caps removed. Valve free
35	6	15	5600	5400	2/3	50	4650	5*	16	6	14	Ice caps removed. Inside front of scoop almost completely closed ahead of valve. Valve free
36	6	10	5400	5400	1/3	100	4250	− 4*	21	6	14	Ice caps removed. Very little ice formed ahead of valve. Valve free

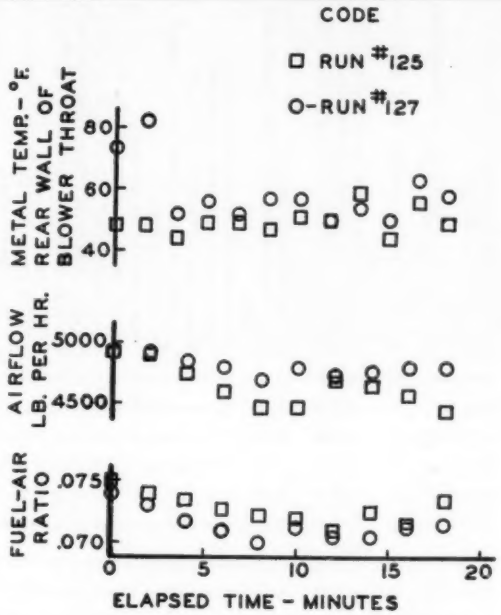

FIG. 18. Effect of oil-in temperature on rate of ice formation with X-bar fuel distributor.

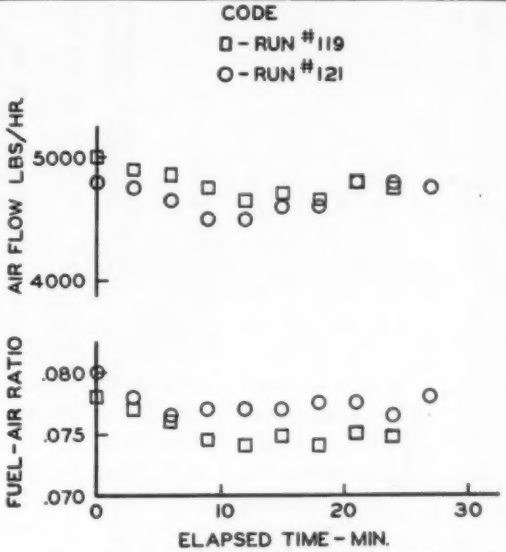

FIG. 19. Effect of fuel temperature on rate of ice formation with X-bar fuel distributor.

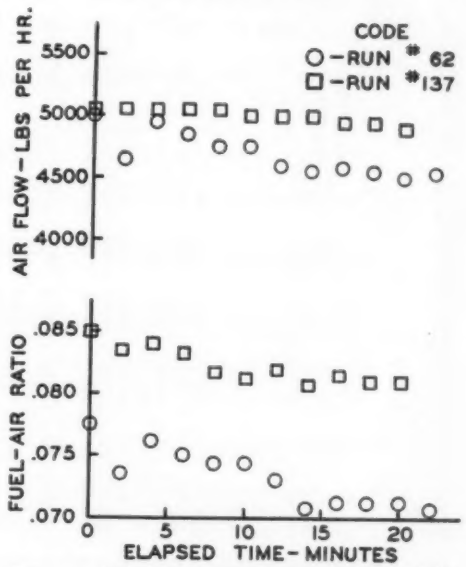

FIG. 20. Effect of scoop pressure on rate of ice formation with X-bar fuel distributor.

Impact Ice

1. Impact ice may seriously affect engine operation for air temperatures below 32°F. where snow, sleet or sub-cooled liquid exists in the atmosphere.

2. A satisfactory scoop design was developed for operation under impact icing conditions.

3. The first design of scoop tested was not satisfactory for operation under impact icing conditions.

4. The most dangerous impact ice is formed on the metering elements of the carburetor.

5. Impact ice may form at or on the entrance screen, the scoop entrance without a screen, the preheater valve, the scoop wall, the scoop elbow, the carburetor screen and the metering elements of the carburetor.

6. A streamlined deflector is an effective icing shield.

Throttle Ice

1. Serious throttle ice may form at carburetor air temperatures between 32°F. and 37°F. with a relative humidity greater than 100 per cent and throttle openings of less than 45 degrees.

Fuel Evaporation Ice

1. Using the X-bar fuel nozzle, no dangerous ice will form with relative humidities of less than 50 per cent or with carburetor air temperatures above 50°F. with any other possible conditions.

EXPLANATORY DATA—TABLE 3

* Dry ice in tunnel.

† X = Distance from trailing edge of shield to scoop entrance.

Systems: (1) original scoop and balanced valve; (2) System No. 1 with entrance screen; (3) original scoop and flapper valve; (4) System No. 3 with plenum chamber; (5) System No. 4 with entrance screen and icing shield; (6) redesigned scoop.

Valve positions: (C) closed, full cold; (O) open, full hot; (1/3) 1/3 open; (2/3) 2/3 open.

Spray systems: (1) small jets—center position; (2) medium jets—center position; (3) large jets—center position; (4) small jets—lower position; (5) medium jets—lower position; (6) large jets—lower position.

TABLE 4

Summary of Engine and Carburetor Icing Tests Using X-Bar Fuel Distributor

Run No.	Length of Run, min.	C.A.T., °F.	Water Conc. (1)	Carb. Air Flow, lb./hr.		Throttle Angle (2)	Initial Throttle Pres. Drop, In.-Hg	Scoop Pres., In.-Hg Abs.	Mixture Setting (3)	F/A Ratio		Oil-in Temp., °F.	Fuel Temp., °F.	Blower Throat Metal Temp., °F.		Engine Operation Effect (4)			Amount Ice Formed (5)						Remarks
				Max.	Min.					Max.	Min.			Max.	Min.	Air Flow	F/A	Distribution	Impact Ice	Butterfly	Fuel Nozzle	X-Bars	Blower Throat	Vanes	
8	20	80	100%	5150	5050	32	12.3	28.05	AL	.0802	.0786	152L	35	..	..	0	0	1	:	:	:	:	:	:	No inspection
10	21	70	100%	5170	5130	32	12.4	28.05	AL	.0805	.0781	153L	34	42	40	0	0	2	0	0	3	2	3	2	Throat ice was on traverse bar
12	76	60	100%	5170	4990	33	10.9	27.90	AL	.0800	.0746	155L	36	..	..	0	1	1	0	0	3	2	0	3	Slushy ice
14	38	50	100%	5080	5050	32	11.4	27.90	AL	.0781	.0759	154L	36	44.5	39	0	0	1	0	0	2	2	0	2	Slushy ice
16	60	40	100%	5120	5030	32	9.85	27.70	AL	.0817	.0748	155L	39	42.5	39.5	0	1	1	0	0	2	2	0	2	Slushy ice
18	44	32	100%	5150	4550	33	10.4	29.60	AL	.0831	.0770	156L	41	37	28.5	1	2	1	0	2	2	2	0	2	Clear, solid ice on butterfly de-icer plates
20	36	12	100%	4750	4000	27	8.4	27.70	AL	.0706	.0535	155L	38	35.5	28.5	2	2	1	0	1	1	1	1	1	Heavy frost. Coils frosting caused drop in air flow
22	36	−17	100%	5000	4300	30	9.3	26.95	AL	.0684	.0624	157L	45	−3	−14	0	2	0	0	1	1	1	1	1	Heavy frost. F/A change due fuel flow trouble
24	60	80	.0086W	5000	5000	34	8.8	27.90	AL	.0830	.0800	155L	40	63	57	0	0	1	0	0	2	1	0	1	
26	48	70	.0086W	5000	5000	33	9.4	27.90	AL	.0810	.0798	154L	39	60	53	0	0	1	0	0	2	2	2	2	
28	45	60	.0086W	4900	4800	32	9.4	27.95	AL	.0781	.0764	157L	40	54.5	47	0	0	2	0	0	3	3	3	3	Throat ice probably fell from X-bars
30	29	49	.0086W	5000	4750	29	9.0	29.60	AL	.0800	.0757	155L	43	48.5	49	1	1	2	0	0	3	3	3	3	Throat ice probably fell from X-bars
32	66	40	.0086W	5000	3350	27	10.7	28.25	AL	.0785	.0510	157L	42	53	36	3	3	3	0	0	3	3	3	3	Ice built from vanes back to rear wall
34	14	34	.0086W	5000	2350	30	11.2	28.50	AL	.0807	.0682	154L	33	50.5	39	3	2	3	0	3	2	2	3	2	Lower part of throat nearly blocked
36	35	11	.0086W	5120	3750	26	8.65	27.80	AL	.0735	.0499	154L	43	34.5	28	3	3	2	3	1	2	2	0	2	Ice built down from X-bar against throat rear wall
38	23	−18	.0086W	4950	3000	25	9.15	28.05	AL	.0650	.0358	155L	33	34.5	17	3	3	2	3	2	2	2	0	2	Impact tubes clogged
40	50	79	.0086S	5180	5120	35	8.6	27.80	AL	.0803	.0779	155L	40	64	62.5	0	0	1	0	0	2	2	0	2	Impact tubes clogged
42	34	71	.0086S	5120	5120	33	9.1	28.10	AL	.0793	.0788	154L	43	67.5	65.5	0	0	2	0	0	3	3	2	2	Slushy ice
44	31	60	.0086S	5100	5030	33	9.2	28.00	AL	.0792	.0769	154L	43	49	32.5	0	0	2	0	0	3	2	2	2	Slushy ice would build on X-bars then break off
46	62	51	.0086S	5120	4730	28	9.2	28.20	AL	.0782	.0731	155L	42	53	46	1	1	3	0	0	3	3	3	3	Ice formations smoother than with free water inject
48	50	40	.0086S	5030	4400	32	8.9	27.95	AL	.0778	.0705	156L	38	55	40.5	2	2	2	0	0	3	3	3	3	Ice from X-bars and vanes joined
50	28	36	.0086S	5180	2840	30	10.8	29.10	AL	.0763	.0614	156L	40	61	39.5	3	3	3	0	3	3	3	0	3	Impeller entrance nearly blocked
52	60	38	50%	5030	5030	28	9.2	27.80	AL	.0753	.0748	156L	40	37.5	31	0	0	0	0	0	0	0	0	0	One butterfly covered much more than other one
54	42	40	100%	5000	4650	29	9.7	28.00	AL	.0744	.0705	155L	37	46.5	39.5	1	1	1	0	1	2	2	0	2	No ice on inspection
56	70	40	.0086W	4950	2000	33	8.6	27.45	AL	.0915	.0692	156L	42	68.5	41	3	3	3	0	0	3	3	3	3	Some fuel must have back-washed to butterfly
58	57	40	.0172W	5060	2780	32	8.4	29.10	AL	.0836	.0667	154L	41	57.5	36.5	3	3	3	0	0	3	3	3	3	Throat almost completely clogged
60	30	41	.088W	5060	2420	32	8.5	29.10	AL	.0825	.0705	156L	41	58	47	3	3	3	0	0	3	3	3	3	Throat almost completely clogged
62	22	40	.0086S	5050	4500	27	9.4	27.80	AL	.0775	.0707	154L	39	50	40	2	2	3	0	0	3	3	3	3	Throat almost completely clogged
64	24	40	.0172S	5080	4560	30	7.4	28.00	AL	.0768	.0712	153L	40	49	45.5	2	2	3	0	0	3	3	3	3	Impeller entrance above vanes clogged
66	65	60	50%	5040	5040	31	9.0	27.70	AL	.0794	.0773	153L	39	52.5	46	0	0	0	0	0	2	2	2	2	Ice slushy with fuel
68	45	60	100%	5000	4900	31	9.1	27.80	AL	.0798	.0769	155L	45	55.5	50.5	0	0	1	0	0	2	2	2	3	Thin layers of ice as noted
70	45	60	.0086W	5060	5060	32	9.0	27.70	AL	.0808	.0790	157L	40	55	48.5	0	0	2	0	0	2	2	3	3	Throat ice on fuel pipe
72	42	61	.0172S	5150	5060	32	9.1	27.85	AL	.0806	.0786	156L	40	52	47	1	0	2	0	0	2	2	3	3	Chunks broke off X-bars while running
74	42	60	.088W	5000	4900	32	9.0	27.85	AL	.0821	.0795	155L	38	63.5	54	1	0	3	0	0	3	3	3	3	Chunks broke off X-bars while running
76	51	60	.0086S	5220	5140	31	9.15	29.05	AL	.0803	.0781	155L	38	53	45	1	0	1	0	0	2	2	2	2	Ice more solid than for No. 70 and No. 72
78	25	60	.0172S	5240	5120	30	8.0	27.95	AL	.0833	.0705	156L	27	55	49	1	2	2	0	0	3	3	3	3	Slushy ice
80	13	5	.0172W	5100	2700	27	8.9	24.70	AL	.0670	.0340	154L	50	44	23.5	3	3	3	3	2	1	1	2	2	More ice than No. 76
84	23	80	.0086W	9720	9720	FT	2.2	26.30	FR	.1178	.1170	163L	41	70.5	70	0	0	0	0	0	0	0	0	0	Impact tubes clogged
																									No ice on inspection

86	57	70	.0086W	9850	9700	FT	2.15	26.25	FR	.1185	.1102	157L	43	57	56.5	1	1	1	0	0	1	0	3	3	Small formation between fuel pipe and rear wall
88	48	60	.0086W	10000	9550	FT	2.1	26.65	FR	.1123	.1027	156L	35	57	44.5	2	2	1	0	0	3	3	3	3	Chunks ice broke off X-bars while running
90	20	50	.0086W	9950	9050	FT	1.8	26.60	FR	.1270	.1223	154L	40	50.5	42	2	2	3	0	0	3	3	3	3	Formation on vanes larger than No. 88
92	21	39	.0086W	9750	5700	FT	1.7	27.70	FR	.1238	.1040	154L	39	53	35	3	3	3	0	0	1	2	3	3	Ice probably broke from X-bars before inspection
94	9	32	.0086W	9800	4800	FT	2.25	27.50	FR	.1200	.0330	162L	24	56	38.5	3	3	3	2	2	3	3	3	3	Throat clogged. Clear ice on carburetor screen
96	37	60	.0086W	2570	2570	16	9.15	27.80	AL	.0803	.0790	159L	43	80	74	0	0	1	0	0	3	2	0	0	No ice on vanes
98	47	49	.0086W	2470	2470	15	9.0	27.75	AL	.0843	.0799	162L	40	78.5	74.5	0	1	3	0	0	3	3	2	2	Ice built up from X-bars
100	27	39	.0086W	2420	2120	15	9.0	28.20	AL	.0856	.0809	160L	44	84.5	76	1	1	3	0	0	3	3	2	2	Ice broke off just before inspection
102	29	32	.0086W	2380	990	16	10.1	28.10	AL	.1192	.0757	158L	29	124	96	3	3	1	0	3	2	2	0	0	Clear ice on upper edge of butterfly
104	37	11	.0086W	2640	1900	30	9.4	28.80	AL	.0740	.0527	154L	53	72.5	57	3	3	1	3	2	0	1	0	0	Screen covered with snow
106	26	40	.0086S	4970	4440	29	7.3	28.00	AL	.0762	.0697	154L	40	48.5	30.5	3	2	3	0	0	3	3	3	3	Impeller entrance nearly blocked
108	17	40	.0086S	4930	4320	30	7.4	28.00	AR	.0865	.0819	156L	41	48	36	3	2	3	0	0	3	3	3	3	Same type ice formation as No. 106
110	17	40	.0086S	4980	4450	30	7.2	27.80	FR	.0938	.0872	154L	40	49	40.5	3	2	3	0	0	3	3	3	3	Same type ice formation as No. 106 and No. 108
112	17	60	.0086S	4980	4920	29	8.1	27.90	AL	.0787	.0770	158L	38	51	46.5	1	1	3	0	0	3	3	2	2	Large chunk ice on fuel nozzle
114	18	60	.0086S	4950	4880	28	7.9	27.70	AR	.0892	.0860	154L	41	52	46	1	1	3	0	0	3	3	2	2	Same type ice formation as No. 112
116	14	60	.0086S	4900	4850	28	7.7	27.60	FR	.0936	.0909	155L	40	53	47	1	1	3	0	0	3	3	2	2	Same type ice formation as No. 112 and No. 114
119	29	41	.0086S	5000	4620	29	7.5	27.80	AL	.0779	.0739	156L	61	48.5	44	1	2	3	0	0	3	3	3	3	Less ice than No. 121
121	32	41	.0086S	4800	4470	28	7.1	28.10	AL	.0800	.0764	154L	40	48.5	38.5	1	1	3	0	0	3	3	3	3	More ice than No. 119
125	20	40	.0086S	4920	4480	27	8.5	27.90	AL	.0748	.0710	140L	40	50.5	42.5	2	1	3	0	0	3	3	3	3	Ice built down from nozzle against rear wall
127	20	40	.0086S	4920	4700	28	9.0	27.70	AL	.0739	.0702	180H	40	58	47.5	1	1	3	0	0	3	3	3	3	Less ice than No. 125
129	21	40	.0086S	5060	4780	FT	0.3	27.65	AL	.0783	.0720	154H	37	66	61	1	2	3	0	0	3	3	3	3	Small amount ice on nozzle, large amount on vanes
131	14	40	.0086S	5080	4450	34	4.6	28.10	AL	.0750	.0703	158H	40	50	40.5	2	2	3	0	0	3	3	3	3	Ice formations inter. between No. 129 and No. 131
133	22	40	.0086S	5050	4780	24	10.3	27.85	AL	.0780	.0750	157H	39	60	44	1	1	3	0	0	3	3	3	3	Large amount ice on nozzle, less on vanes
135	21	41	.0086S	5140	4050	FT	0.5	25.80	AL	.0821	.0758	155H	40	78.5	70.5	3	2	3	0	0	3	3	3	3	Impeller entrance nearly plugged
137	21	40	.0086S	5060	4900	48	2.85	19.90	AL	.0850	.0807	172H	40	84.5	72	1	1	2	0	0	3	2	3	3	Ice built around fuel pipe
139	30	60	.0086S	4900	4850	48	2.9	20.05	AL	.0860	.0831	185H	39	89	80	1	1	3	0	0	3	3	3	3	More ice in throat than for No. 112
141	21	40	.0086S	6920	5900	37	8.45	27.45	AL	.0954	.0850	162H	40	53.5	47	3	2	3	0	0	2	3	3	3	More ice than No. 145, less than No. 143
143	18	40	.0086S	4980	4420	37	3.75	27.75	AL	.0773	.0728	158H	42	53	42.5	2	2	3	0	0	3	3	3	3	More ice in throat than No. 141 and No. 145
145	21	40	.0086S	3300	3240	37	1.65	27.30	AL	.0785	.0753	165H	42	63.5	58	1	1	3	0	0	3	3	3	3	Less ice than No. 141 and No. 143
220	9	32	.0025	4650	2410	29	8.6	28.90	AL	.0778	.0756	155L	41	..	..	3	2	3	0	3	2	3	0	1	No ice in blower throat
221	18	13	.0025	4940	3030	28	8.0	24.50	AL	.0522	.0355	154L	41	..	..	3	3	3	3	3	3	3	0	2	Impact tubes clogged
222	24	4	.0025	5030	1920	27	8.65	25.92	AL	.0723	.0300	155L	42	..	..	3	2	2	3	3	1	2	0	0	Impact tubes clogged
223	21	80	50%	5100	5140	30	7.55	27.80	AL	.0784	.0769	155L	40	52	49	0	0	1	0	0	1	0	0	1	Trace on bottom of vanes
224	21	60	25%	5140	5150	30	7.95	27.80	AL	.0798	.0767	155L	40	44.5	41.5	0	0	0	0	0	0	0	0	0	No ice on inspection
225	24	80	25%	5160	5210	30	7.7	27.80	AL	.0804	.0777	155L	40	52	47	0	0	0	0	0	0	0	0	0	No ice on inspection
226	18	90	25%	5140	5150	30	7.65	27.80	AL	.0782	.0771	155L	41	55	51	0	0	0	0	0	0	0	0	0	No ice on inspection
227	16	20	75%	5060	5060	30	8.35	27.80	AL	.0775	.0762	155L	40	20	19.5	0	0	0	1	1	1	1	0	0	Trace of frost on throttle, etc.
228	21	32	75%	5100	5100	30	8.3	27.80	AL	.0780	.0765	155L	40	33.5	31.5	0	0	0	1	1	1	1	0	0	Trace of frost on throttle, etc.
229	16	−17	75%	5370	5370	30	8.9	27.80	AL	.0662	.0646	155L	40	16	14	0	0	0	1	1	1	1	0	0	Trace of frost on throttle, etc.
230	17	90	100%	5040	5040	30	7.2	27.80	AL	.0825	.0784	155L	40	60.5	59	0	0	0	0	0	0	0	0	0	No ice on inspection
231	17	90	.0172W	5010	5020	30	7.3	27.80	AL	.0832	.0792	155L	40	62.5	61	0	0	0	0	0	0	0	0	0	No ice on inspection
232	17	80	.088W	5140	5140	30	7.75	27.80	AL	.0788	.0768	155L	40	65.5	65	0	0	0	0	0	0	0	0	0	No ice on inspection
233	18	40	75%	4960	4970	30	8.0	27.80	AL	.0800	.0765	155L	40	37.5	36	0	0	1	0	0	1	0	0	1	Trace on bottom of vanes

(1) Water concentration in % R.H., or in lbs. water in excess of 100% R.H. per lb. of dry air. W and S indicate water or steam injection, respectively.
(2) 0° corresponds to fully closed throttle, while full throttle is 77°.
(3) AL = auto lean (F/A approx. 0.082 to 0.072); AR = auto rich (F/A approx. 0.106 to 0.103); FR = full rich.
(4) The severity of the effect on engine operation of a change in air flow, F/A, or distribution, caused by icing, is shown by the following: (0) no effect; (1) slight effect; (2) marked effect; (3) dangerous effect.
(5) The amount of ice formed on the various parts is indicated by the following: (0) no ice; (1) up to $1/16''$ of ice; (2) $1/16''$ to $1/4''$; (3) thicker than $1/4''$.

TABLE 5

Summary of Engine and Carburetor Icing Tests Using New Type Fuel Nozzle

Run No.	Length of Run, min.	C.A.T., °F.	Water Conc. (1)	Carb. Air Flow, lb./hr.		Throttle Angle (2)	Initial Throttle Pres. Drop, In.-Hg	Scoop Pres., In.-Hg Abs.	Mixture Setting (3)	F/A Ratio		Oil-in Temp., °F.	Fuel Temp., °F.	Blower Throat Metal Temp., °F.		Engine Operation Effect (4)			Amount Ice Formed (5)						Remarks
				Max.	Min.					Max.	Min.			Max.	Min.	Air Flow	F/A	Distribution	Impact Ice	Butterfly	Fuel Nozzle	X-Bars	Blower Throat	Vanes	
7	20	78	100%	5120	5070	33	11.6	27.50	AL	.0838	.0822	156L	38	...	...	0	0	0	..	..	..	..	..	..	No inspection
9	21	79	100%	5100	5100	32	12.0	28.40	AL	.0847	.0839	155L	33	71.5	71.5	0	0	0	..	..	..	..	..	..	No inspection
11	95	60	100%	5150	5100	33	10.5	27.70	AL	.0820	.0807	153L	35	...	...	0	0	0	..	..	..	..	..	..	No inspection
13	32	51	100%	5100	5050	29	12.2	27.70	AL	.0822	.0801	157L	37	66	63.5	0	0	0	0	0	0	0	0	0	No ice on inspection
15	40	41	100%	5000	5000	29	10.3	27.80	AL	.0789	.0777	155L	39	68.5	64	0	0	0	0	0	0	0	0	0	No ice on inspection
17	40	32	100%	5100	4900	31	9.7	27.30	AL	.0820	.0796	155L	32	66	57	0	0	0	0	0	0	0	0	1	Bottom of right vane
21	51	−17	100%	5050	4900	28	10.4	27.48	AL	.0755	.0723	156L	35	31	24.5	0	0	0	0	1	1	1	1	1	Trace of frost
23	50	80	.0086W	5050	5050	34	8.7	27.65	AL	.0850	.0825	155L	45	81.5	80.5	0	0	0	..	..	..	..	..	..	No inspection
25	55	69	.0086W	5050	5000	33	9.2	27.85	AL	.0853	.0840	156L	37	73	71	0	0	0	..	..	..	..	..	..	No inspection
27	60	63	.0086W	5050	4950	34	9.4	27.70	AL	.0840	.0814	155L	36	72.5	70.5	0	0	0	..	..	..	..	..	..	No inspection
29	35	50	.0086W	5050	5050	30	9.7	27.75	AL	.0800	.0796	156L	42	59	56	0	0	0	0	0	0	0	0	0	Small chunk ice on vane thermocouple wire
31	75	41	.0086W	5150	5000	27	10.0	28.05	AL	.0785	.0771	155L	36	52	48.5	0	0	0	0	0	0	0	0	2	1/8" on impeller edge of vanes
33	35	32	.0086W	5000	2900	30	11.1	28.10	AL	.0804	.0705	156L	46	78	48.5	3	2	1	0	3	0	0	0	2	Clear ice on butterfly. Built up on impeller side of vanes
35	39	10	.0086W	5100	3450	25	8.6	27.25	AL	.0590	.0310	156L	39	53.5	47	3	3	2	3	1	1	1	0	2	Impact tubes clogged
37	41	−21	.0086W	5050	3700	25	8.1	28.85	AL	.0785	.0350	155L	43	38.5	24	3	3	2	3	1	1	1	0	2	Impact tubes clogged
39	45	79	.0086S	5150	5150	35	8.5	27.45	AL	.0826	.0804	155L	43	89.5	83.5	0	0	0	..	..	..	..	..	..	No inspection
41	50	70	.0086S	5150	5150	33	9.1	27.80	AL	.0828	.0807	158L	41	75	74	0	0	0	..	..	..	..	..	..	No inspection
43	30	60	.0086S	5150	5050	33	9.2	27.75	AL	.0822	.0800	156L	40	72	67	0	0	0	..	..	..	..	..	..	No inspection
45	60	49	.0086S	5200	4100	30	8.1	25.85	AL	.0858	.0720	156L	42	69	59	0	0	0	0	0	0	0	0	1	Small piece on vane thermocouple wire. Coils frosted
47	46	41	.0086S	5200	5150	32	9.0	27.85	AL	.0795	.0784	155L	37	55.5	54.5	0	0	0	0	0	0	0	0	2	1/8" ice built up on impeller side of vanes
49	50	35	.0086S	4450	3700	32	11.1	28.95	AL	.0773	.0640	156L	42	54	27	2	3	3	0	3	0	3	0	2	Large formation of clear ice between X-bars and butterflies
51	60	44	50%	5050	5000	28	9.2	27.80	AL	.0770	.0759	156L	37	79.5	70	0	0	0	0	0	0	0	0	0	No ice on inspection
53	30	41	100%	5050	4950	29	9.2	27.80	AL	.0773	.0763	157L	37	70.5	69.5	0	0	0	0	0	0	0	0	0	No ice on inspection
55	115	40	.0086W	5000	4750	32	8.3	27.85	AL	.0773	.0712	156L	36	49.5	36	0	0	0	0	0	0	0	0	2	Ice chunk on vane thermocouple wire
57	55	40	.0172W	5050	4650	33	8.0	27.90	AL	.0808	.0782	156L	42	48.5	36.5	1	2	3	0	0	0	0	3	3	Leak in fuel pipe caused large ice formation on vanes
59	60	41	.088W	5050	4900	28	8.5	28.55	AL	.0810	.0755	155L	41	37	48.5	0	1	2	0	0	0	0	3	1	Ice behind fuel pipe caused by leak
61	100	41	.0086S	5150	4650	27	7.8	27.80	AL	.0813	.0729	154L	43	52	38	1	2	3	0	0	0	0	3	3	Leak in fuel pipe caused large ice formation
63	40	40	.0172S	5100	5050	30	7.5	27.83	AL	.0767	.0754	155L	42	52	49.5	0	0	0	0	0	0	0	0	2	Small chunk ice on vane thermocouple wire

65	75	61	50%	5050	5050	31	9.1	27.70	AL	.0805	.0790	155L	39	83.5	61	0	0	0	0	0	0	0	0	0	No ice on inspection
67	50	61	100%	5050	5050	31	9.0	27.70	AL	.0807	.0794	154L	42	83.5	81	0	0	0	0	0	0	0	0	0	No ice on inspection
69	50	60	.0086W	5050	5000	32	8.6	27.70	AL	.0833	.0806	155L	42	43.5	39.5	0	0	1	0	0	0	0	3	3	Leaking fuel pipe caused ice
71	55	60	.0172W	5150	5050	30	9.0	27.80	AL	.0825	.0806	155L	39	44.5	38.5	0	0	1	0	0	0	0	2	0	Leaking fuel pipe caused ice
73	75	60	.088W	5050	5000	28	8.9	27.85	AL	.0822	.0812	157L	41	70	69	0	0	0	0	0	0	0	0	0	No ice on inspection
75	60	60	.0086S	5250	5200	21	9.0	27.85	AL	.0830	.0809	158L	34	70	66.5	0	0	0	0	0	0	0	0	0	No ice on inspection
77	55	60	.0172S	5000	4950	30	8.3	28.27	AL	.0789	.0750	151L	41	65	63	0	0	0	0	0	0	0	0	0	No ice on inspection
79	14	10	.0172W	5050	4700	27	7.9	27.50	AL	.0555	.0225	147L	45	56	48.5	2	3	3	3	1	1	1	0	2	Impact tubes clogged
83	25	80	.0086W	9650	9550	FT	2.2	26.35	FR	.1145	.0953	167L	35	103.5	95	0	2	0	0	0	0	0	0	0	Fuel pressure very erratic
85	35	70	.0086W	9900	9850	FT	2.2	26.25	FR	.1310	.0788	164L	39	82	79.5	0	2	0	0	0	0	0	0	0	Fuel pressure very erratic
87	45	61	.0086W	10015	10015	FT	2.1	26.20	FR	.1047	.0953	160L	49	72	67	0	1	0	0	0	0	0	0	0	Fuel pressure very erratic
89	28	50	.0086W	10015	10005	FT	1.7	26.25	FR	.1040	.1005	154L	41	62	60.5	0	0	0	0	0	0	0	0	0	Fuel pressure very erratic
91	32	40	.0086W	9900	9400	FT	1.8	26.50	FR	.1090	.1040	155L	39	52	50.5	0	0	0	0	0	0	0	0	0	No ice on inspection. Fuel pressure steadier
93	21	32	.0086W	10000	8200	FT	2.5	26.70	FR	.1100	.0375	157L	37	56	46	3	3	1	2	2	1	1	0	2	No ice on inspection
95	75	60	.0086W	2550	2550	16	9.2	27.80	AL	.0935	.0808	165L	48	84	81	0	2	0	0	0	0	0	0	0	Clear ice on impact tubes and boost venturi
97	50	50	.0086W	2500	2450	14	9.1	27.85	AL	.0842	.0803	163L	44	87	81.5	0	0	0	0	0	0	0	0	0	No inspection. Fuel flow erratic
99	50	40	.0086W	2450	2450	15	8.7	27.70	AL	.0840	.0799	159L	45	78	75	0	0	0	0	0	0	0	0	0	No ice on inspection
101	45	30	.0086W	2550	2400	16	9.0	27.75	AL	.0821	.0787	159L	49	71	69.5	0	0	0	0	0	0	0	0	0	No ice on inspection
103	28	10	.0086W	2450	1500	16	8.7	25.30	AL	.0950	.0625	159L	42	108	68	3	3	1	3	1	0	0	0	0	Screen covered with snow
105	28	40	.0086S	5050	5000	29	8.1	27.85	AL	.0750	.0731	155L	39	52	50	0	0	0	0	0	0	0	0	2	Ice on vane thermocouple wires
107	30	40	.0086S	5050	4950	29	7.6	28.88	AR	.0845	.0834	155L	39	52	49.5	0	0	0	0	0	0	0	0	2	Ice on vane thermocouple wires
109	30	40	.0086S	4900	4800	30	7.8	27.95	FR	.0863	.0843	154L	41	51.5	50	0	0	0	0	0	0	0	0	2	Ice on vane thermocouple wires
111	28	60	.0086S	5050	5050	29	8.3	27.95	AL	.0803	.0766	154L	35	65	62.5	0	0	0	0	0	0	0	0	1	Small piece ice on vane thermocouple wire
113	32	60	.0086S	5000	4900	28	8.0	27.83	AR	.0866	.0799	155L	41	56.5	56	0	0	0	0	0	0	0	0	1	Small piece ice on vane thermocouple wire
115	28	61	.0086S	4900	4900	28	7.9	27.50	FR	.0923	.0892	155L	41	67	64.5	0	0	0	0	0	0	0	0	1	Small piece ice on vane thermocouple wire
118	36	41	.0086S	5150	5100	29	7.6	27.75	AL	.0772	.0754	156L	61	55	52.5	0	0	0	0	0	0	0	0	1	Small piece ice on vane thermocouple wire
120	36	40	.0086S	5050	5050	29	7.4	27.80	AL	.0772	.0755	155L	40	52.5	51	0	0	0	0	0	0	0	0	1	Small piece ice on vane thermocouple wire
122	20	40	.0086S	5100	5000	29	8.0	27.80	AL	.0775	.0300	156L	25	53	49.5	0	1	1	0	0	0	0	0	2	Fuel flow erratic. Thin layer ice on vanes
124	28	40	.0086S	5050	4950	27	8.5	27.80	AL	.0762	.0733	142L	37	52	50	0	0	1	0	0	0	0	0	2	1/32" layer ice in places on vanes
126	28	40	.0086S	4950	4900	26	9.9	27.85	AL	.0743	.0714	180H	40	56.5	55	0	0	1	0	0	0	0	0	2	More ice than No. 124, in same locations
128	28	41	.0086S	5100	5050	FT	0.4	27.40	AL	.0773	.0752	154H	39	65	62.5	0	0	1	0	0	0	0	0	2	Ice built up 3/16" on impeller edge of vanes
130	28	40	.0086S	5100	5050	34	4.8	27.85	AL	.0775	.0753	165H	34	...	...	0	0	1	0	0	0	0	0	2	Less ice than No. 128
132	28	40	.0086S	5150	5100	24	10.5	27.53	AL	.0773	.0760	171H	39	57.5	56.5	0	0	0	0	0	0	0	0	1	Less ice than No. 130
134	28	39	.0086S	5250	5200	FT	0.3	26.00	AL	.0825	.0804	162H	43	73	65	0	0	2	0	0	0	0	0	2	Ice built up 1/4" on impeller edge of vanes
136	28	40	.0086S	5150	5150	48	2.3	19.55	AL	.0875	.0815	190H	40	97	90	0	0	1	0	0	0	0	0	1	Less ice than No. 126
138	28	60	.0086S	4900	4800	49	2.7	19.45	AL	.0848	.0822	190H	40	117	111.5	0	0	0	0	0	0	0	0	0	No ice on inspection
140	28	40	.0086S	7150	7050	37	8.8	27.48	AL	.0930	.0914	166H	38	57	54	0	0	1	0	0	0	0	0	1	Trace ice on impeller edge of vanes
142	32	40	.0086S	5150	5100	37	3.9	27.75	AL	.0760	.0730	161H	40	60	56.5	0	0	1	0	0	0	0	0	2	More ice than for No. 140 and No. 144
144	28	40	.0086S	3400	3200	37	1.7	27.40	AL	.0830	.0765	168H	40	69	64.5	0	0	1	0	0	0	0	0	1	Less ice than for No. 140

(1) Water concentration in % R.H., or in lbs. water in excess of 100% R.H. per lb. of dry air. W and S indicate water or steam injection, respectively.
(2) 0° corresponds to fully closed throttle, while full throttle is 77°.
(3) AL = auto lean (F/A approx. 0.082 to 0.072); AR = auto rich (F/A approx. 0.106 to 0.103); FR = full rich.
(4) The severity of the effect on engine operation of a change in air flow, F/A, or distribution, caused by icing, is shown by the following: (0) no effect; (1) slight effect; (2) marked effect; (3) dangerous effect.
(5) The amount of ice formed on the various parts is indicated by the following: (0) no ice; (1) up to 1/16" of ice; (2) 1/16" to 1/4"; (3) thicker than 1/4".

TABLE 6
Prevention of Ice with Preheat

Run No.	Temp. at Carb., °F. Start	End	Cold Air Flow, lb./hr. Start	End	Hot Air Flow, lb./hr. Start	End	Water Concentration* of Combined Flow	Throttle Position	Heater Valve Position	Nozzle	Amount of Ice X-Bar	Blower Throat	Vanes
153	49	49	..	..	8910	8650	100%	Full	O	2	2	0	3
154	61	60	..	..	9000	8930	100%	Full	O	0	0	0	2
155	78	80	2530	2530	2640	2680	100%	Part	1/2	1	1	0	0
156	70	67	2620	2520	2520	2520	100%	Part	1/2	2	1	0	2
157	59	59	2440	2440	2710	2710	100%	Part	1/2	2	1	0	2
158	51	51	2310	2250	2650	2580	100%	Part	1/2	2	2	0	2
160	79	81	2570	2530	2500	2480	.0086W	Part	1/2	2	0	0	0
161	70	69	2300	2300	2640	2640	.0086W	Part	1/2	2	1	0	1
162	61	58	2310	2280	2550	2590	.0086W	Part	1/2	1	1	0	0
163	53	49	1960	2300	3030	2300	.0086W	Part	1/2	1	1	0	2
165	91	94	2200	2230	2500	2480	.0086W	Part	3/4	0	0	0	0
166	91	90	2150	2150	2420	2420	.0086W	Part	3/4	0	0	0	0

* % R.H., or free water in excess of 100% R.H.—lb./lb. of dry air (W = water, or S = steam).
Heater valve position: (O) full hot; (1/2) 1/2 open; (3/4), 3/4 open.
Constants for all runs: scoop pressure, 27.8″ Hg; mixture setting, auto-lean; oil-in temp., 155°F.; clutch setting, low; fuel nozzle, X-bar.
Amount of ice: (0) none; (1) trace of ice formed; (2) no dangerous ice formed; (3) dangerous ice formed.
Average static pressure difference between hot and cold air, 8″ H_2O.
Average temperature difference between hot and cold air, 50°F

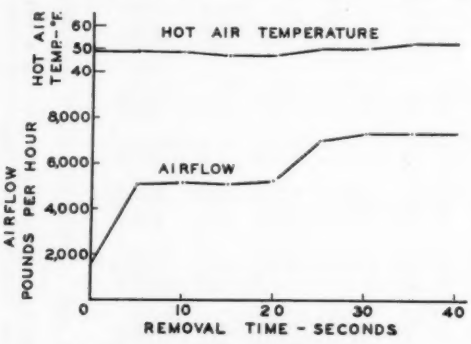

FIG. 21. Ice removal with 50°F. hot air.

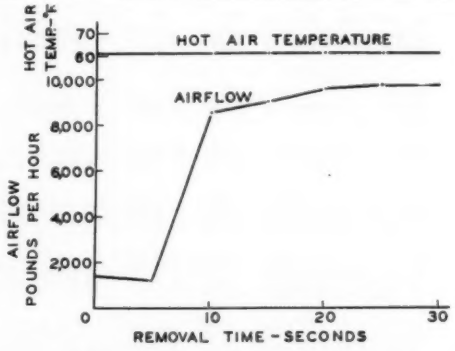

FIG. 22. Ice removal with 60°F. hot air.

TABLE 7
Removal of Ice with Preheat

	Icing Condition									Preheat Conditions					
Run No.	Cold Air Flow, lb./hr. Start	End	Cold Air Temp., °F. Start	End	Water Concentration	Clutch Setting	Throttle Position Start	End	Type of Ice Formed	Water Concentration, %	Hot Air Flow, lb./hr. Start	End	Hot Air Temp., °F. Start	Max.	Time to Restore Full Throttle Air Flow, Sec.
146	5100	3720	42	33	.0086W	Low	30°	30°	1	100	3720	8700	85	81	20
147	4700	3120	41	32	.0086W	Low	30°	30°	1	100	3120	9100	86	90	20
148	4580	4100	39	39	.0086W	Low	30°	30°	2	100	4100	9100	67	84	10
250	4790	2200	38	33	.0086W	Low	30°	Full	1	100	2200	9350	74	80	20
251	4480	2500	36	31	.0086W	Low	30°	Full	1	100	2500	9100	100	102	10
252	5000	1750	37	31	.0086W	Low	30°	Full	1	100	1750	9100	70	76	30
253	4820	1400	38	30	.0086W	Low	30°	Full	3	100	1400	9570	61	61	20
254	4830	2530	36	32	.0086W	Low	30°	Full	1	100	2530	7060	93	93	5†
255*	4890	1050	10	14	.0086S	High	30°	Full	4	100	1050	7300	49	52	30

* New design Pratt & Whitney fuel nozzle used, X-bar nozzle used for all other runs.
† Stopped for 90 sec., flow not restored on starting. Stopped for 150 sec., same result. De-iced with preheat.
Water concentration: % R.H., or free water in excess of 100% R.H.—lb./lb. dry air (W = water, or S = steam).
Constants for icing procedure: scoop pressure, 27.8″ Hg; mixture setting, auto lean; oil-in temp., 155°F.
Types of ice: (1) throttle ice and fuel evaporation ice; (2) fuel evaporation ice; (3) combination of snow and type (1); (4) snow; (5) throttling ice.

TABLE 8

Removal of Ice with Alcohol

Run No.	Icing Condition: Cold Air Flow, lb./hr. Start	Cold Air Flow, lb./hr. End	Cold Air Temp., °F. Start	Cold Air Temp., °F. End	Water Concentration	Throttle Position Start	Throttle Position End	Type of Ice Formed	De-icing Condition: System	Rate of Alcohol Flow, lb./hr.	Time to Restore Full Throttle Air Flow, sec.
181	4660	3630	40	28	.0086W	30°	30°	3	1	10	180
182	4580	2830	38	33	.0086W	30°	Full	1	1	20	130
183	4710	1820	39	31	.0086W	30°	Full	1	1	30	110
184	4860	1180	37	31	.0086W	30°	Full	1	1	40	80
185	4870	1180	34	31	.0086W	30°	Full	1	2	30	70
186	4900	750	33	31	.0086W	30°	Full	1	3	30	70
187	4210	3900	33	31	.0086W	30°	30°	5	3	30	35
188	4750	780	12	−7	.0086S	30°	Full	4	3	30	80
189*	3720	1520	15	16	.0086S	30°	Full	4	...	..	...
190*	4490	4330	34	34	.0086W	30°	30°	5	...	..	...
191*	4710	3610	38	33	.0086W	30°	Full	1	...	..	...
192	4680	1890	41	37	.0086W	30°	Full	2	3	40	90
193	4490	2350	34	33	.0086W	30°	Full	1	3	60	125
194	4790	3010	33	36	.0086W	30°	Full	1	3	80	100
195	5370	940	−20	5	.0086S	30°	Full	4	3	40	90
196	4200	940	0.5	7	.0086S	30°	Full	4	3	60	45
197	4880	770	− 0.5	7	.0086S	30°	Full	4	3	80	35

* Obtained pictures of the three types of ice being removed.

Water concentration: % R.H., or free water in excess of 100% R.H.—lb./lb. dry air (W = water, or S = steam).

Constants for all runs: engine speed 2300 r.p.m.; scoop pressure, 27.8″ Hg; mixture setting, auto-lean; oil-in temp., 155°F.; fuel nozzle, X-bar.

Type of ice: (1) throttle ice and fuel evaporation ice; (2) fuel evaporation ice; (3) combination of snow and (1); (4) snow; (5) throttling ice.

Alcohol systems: (1) two jets at inner radius of elbow; (2) two jets on rear wall opposite (1); (3) combination of (1) and (2).

2. Using the new fuel nozzle no dangerous fuel evaporation ice will form under any possible operating condition.

Prevention and Removal of Ice with Heat

1. The effect on the formation of ice of operation in any preheater valve position with different hot and cold air temperatures at a given average carburetor air inlet condition was negligible. This is surprising in view of the fact that there was considerable variation in temperature of the air at the carburetor inlet in partial preheater valve positions.

2. Preheated carburetor air at a temperature of 60°F., with a relative humidity of 100 per cent or less, is a more rapid de-icing agent than well distributed alcohol flowing at the rate of 80 lbs. per hr. except where the engine has almost stopped. The rate of alcohol removal seemed to be limited by the ability of the supercharger impeller to dispose of large chunks of ice dislodged by the alcohol.

Effect of Ice on Engine Operation

1. Carburetor air flow was effected by impact, throttle, and fuel evaporation ice, and was practically stopped under certain conditions by ice in the blower throat.

2. Fuel to air ratio may be affected by impact icing of the carburetor metering elements or icing of the fuel discharge nozzles.

3. Judging from the character of ice formed on critical parts, mixture distribution was affected by ice on fuel discharge nozzles, X-bar fuel distributor or blower throat.

Design Recommendations

The following are carburetor air scoop design considerations, in the order of their importance: (1) air flow and temperature distribution as it affects carburetor metering, (2) conditions of preheated air, (3) operable preheat valve under impact icing conditions, (4) air pressure at carburetor inlet, (5) drag, (6) weight. The requirements of (2) depend somewhat on the preheat capacity which is required for the prevention and removal of ice. Where it is low such as with the new fuel nozzle tested, removal of free water is very important especially for operation in snow or sleet below strut temperatures of 0°F. For this reason the preheated air should not be taken from behind the engine where it has been heated without effective water separation, because a lower relative humidity will result if the water is separated from the cold air before any heat is added. A relative humidity of less than 50 per cent is desirable and should be easily obtainable. Design changes

recommended for carburetors to improve them from the icing point of view are: (1) Heat the critical parts so that throttle ice will not form under any conditions, (2) simplify the air flow passages as much as possible and eliminate all the irregular obstructions to the air flow, (3) protect carburetor metering elements for impact icing. The cooling capacity of the throttling process on metal parts is so small that it is practical to prevent throttle ice by heating the metal parts.

The following were observations on the engine parts of the induction system: (1) More information is needed on control of the fuel-air mixture distribution. (2) Fuel discharge nozzles should be located as near the impeller entrance as possible. (3) All obstructions in the air flow passage should be eliminated. (4) Possible jamming of ice in any part of the system should be eliminated by draft. (5) Heat transfer should be promoted from hot parts to the induction passage walls. (6) The cooling effect of fuel evaporation on all metal parts was small with the new fuel nozzle, so that it should be practical to supply sufficient heat to eliminate the small amount of ice which was formed there under a few conditions.

Icing and Ice Warning Indicators

Because of the danger from various amounts of ice which may form in a number of places, an ice warning device is preferable to an icing indicator especially if the indicator operates on only localized ice. No satisfactory ice warning device is available at present.

TABLE 9

Fuel Properties of 73 Octane Clear Fuel
Used in induction system icing study

	A.S.T.M. Distillation Temperature, °F	
% Evap.	Calculated Composite 4 Samples	Actual Composite
I.B.P., °F.	113	...
5% at, °F.	140	140
10% at, °F.	147	147
20% at, °F.	158	158
30% at, °F.	168	...
40% at, °F.	179	...
50% at, °F.	188	190
60% at, °F.	199	...
70% at, °F.	208	...
80% at, °F.	218	...
90% at, °F.	239	...
95% at, °F.	259	...
96% at, °F.	269	266
F.B.P., °F.	299	297
Recovery	98.5%	98.0%
Gravity, API		64.9
Reid vapor pressure		6.4

	% Water Solubility	
Fuel Temp., °F.	100 Octane Aviation Gasoline Straight	+15% Aromatics
80	.0085	.0133
32	.0066	.0068
80–32	.0019	.0065

CONCLUSIONS

The following conclusions are justified on the basis of the laboratory tests:

1. Full scale testing in a refrigerated wind tunnel is the best method now available for developing an air scoop which eliminates the hazard of impact icing.

2. The limiting icing conditions and hot air requirements for the prevention and removal of ice in the carburetor and engine parts of the induction system may be determined in the laboratory using full scale equipment including a complete running rear section.

3. With a new type of fuel discharge nozzle, recently developed by Pratt & Whitney Aircraft, there is no operating hazard from fuel evaporation ice forming in the blower throat.

4. With proper operation, preheated air of specified condition is sufficient to eliminate the induction system icing hazards that were investigated.

5. Using an X-bar fuel distributor, ice which dangerously affects manifold pressure, fuel-air ratio and air flow may form with carburetor air temperatures below 55°F. with a relative humidity greater than 90 per cent. Sufficient ice will probably form, affecting mixture distribution, at carburetor air temperatures below 80°F. with a relative humidity greater than 50 per cent.

6. Using the new design of fuel discharge nozzle, throttle ice which dangerously affects manifold pressure and fuel-air ratio may form at carburetor air temperatures below 40°F. with relative humidities greater than 90 per cent.

7. Using the new fuel discharge nozzle, sufficient ice may form on the vane at the impeller entrance to slightly affect mixture distribution at carburetor air temperatures below 50°F. with a relative humidity greater than 100 per cent, but it could be eliminated by further development of the blower throat design.

8. It is possible to design a scoop with a hot air valve that would be fully operable during impact icing conditions.

9. With conventional fuel nozzles and carburetor air temperatures above 32°F., fuel evaporation ice may form to such an extent that carburetor air flow will practically stop.

10. With the systems tested, it is estimated that the engine will not stop running in less than three minutes after the first indication of ice by a drop in manifold pressure, or by rough running of the engine.

11. For safe operation with the X-bar fuel distributor, an alternate air intake system and carburetor air preheater should be provided. For absolutely safe airline operation, the preheater should be capable of supplying 110°F. preheated air with a relative humidity

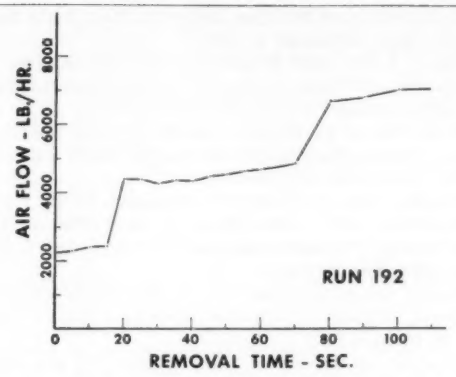

FIG. 23. Ice removal with alcohol flow rate of 40 lbs. per hr.

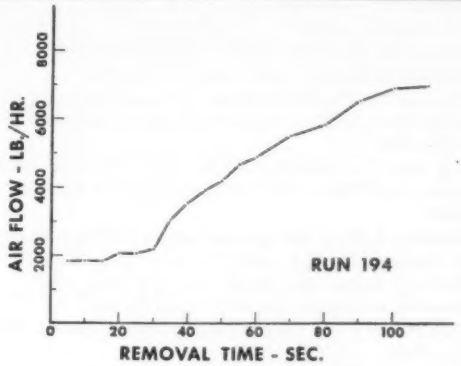

FIG. 24. Ice removal with alcohol flow rate of 80 lbs. per hr.

of 100 per cent or less at a strut temperature of 0°F. and cruising power.

12. For safe operation with the new fuel discharge nozzle, there should be an alternate air intake system capable of supplying hot air at a temperature of 60°F. with a relative humidity of 100 per cent or less at 0°F. strut temperature and cruising power.

13. For operation in snow or sleet at sub-zero temperatures, the relative humidity of preheated air should be 50 per cent or less with a maximum possible amount of free water in the atmosphere, so that the required preheat rise will not be greater than that necessary at 32°F. strut temperature.

14. In the induction system tested, after losing 50 per cent of cruising air flow and with the blower throat practically full of ice formed with the X-bar distributor, 60°F. hot air, with a relative humidity of 100 per cent or less, is a better de-icing agent than well distributed alcohol flowing at a rate of 80 lbs. per hr.

15. The best way to prevent throttle ice is to heat critical parts of the throttle body.

16. The fuel to air ratio or carburetor mixture distribution may be seriously affected by ice before there is an appreciable indication in engine manifold pressure.

17. There is less danger from induction system icing when operating at a high b.m.e.p. because ice is less likely to form and there is more preheat capacity available.

18. Until more operating experience is obtained to correlate these laboratory tests with actual operation, the induction system should be operated so that no ice will form in it.

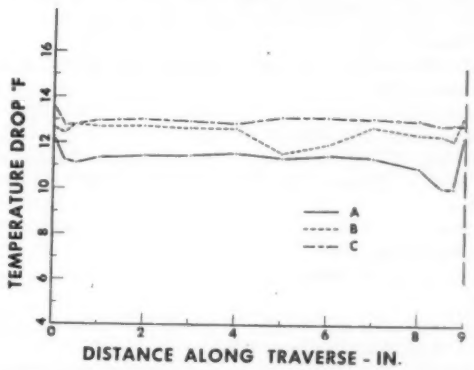

FIG. 25. Temperature traverse indicating alcohol distribution. *A*, *B* and *C*, traverse paths transverse to engine axis.

ACKNOWLEDGMENTS

So many agencies contributed to this study including most of the airlines, the Bendix-Stromberg Carburetor Company, and the Douglas Aircraft Company, that it is impossible to acknowledge all of the assistance. The excellent cooperation and loan of large scale facilities by the York Ice Machinery Corporation and The B. F. Goodrich Company are greatly appreciated.

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Range at Constant Speed

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ABSTRACT

The difference, at constant speed, between the thrust horsepower required at the average weight, as commonly used in range estimates, and the exact "mean effective thrust horsepower" required over the weight interval, as defined in this paper, is investigated. Use of this m.e.t.hp. in estimates of range at constant speed for high-altitude, heavily loaded airplanes is facilitated by curves giving it as a function of t.hp. required at the initial weight, of the ratio of final to initial weights, and of the basic performance parameter $\Lambda_{\sigma 1}$ at the initial weight of the airplane.

THE GENERAL expression for the range in still air (exclusive of climb and descent), in miles, of an airplane with a fuel and oil load such that the gross weight W decreases from an initial weight W_1 lb. to a final weight W_2 lb. is

$$\text{Range} = \int_{W_2}^{W_1} (V\eta/c \text{ t.hp.}) dW \qquad (1)$$

where V is the airplane speed in m.p.h.; η is the overall propulsive efficiency; t.hp. $= \eta$ b.hp., the thrust horsepower required; and c is the specific fuel consumption in lb. per b.hp. per hr.

The calculation of range at constant speed is usually of most practical interest, in which case V can be taken outside of the integral. It is customary, in making preliminary design estimates of ranges at constant speeds, to use constant, average values of efficiency and of specific fuel consumption, or, rather, the average value of the ratio, $\overline{\eta/c}$, inasmuch as η and c are not independent of each other.[1] Further, it is customary to use the thrust horsepower required at the average weight $\overline{W} = (W_1 + W_2)/2$ for the speed and altitude under consideration, *i.e.*, t.hp.$_{\overline{W}}$, so that the range is, approximately,

$$\text{Range} \doteq V\overline{\eta/c}\,(W_1 - W_2)/\text{t.hp.}_{\overline{W}} \qquad (2)$$

In making such estimates for very long-range, high-altitude airplanes, for which the induced drag at cruising speed is a considerable part of the total drag, and for which the weight variation is large (with a consequently large and non-linear variation in thrust horsepower required), it becomes desirable to take into account the weight integral of the reciprocal of t.hp. The range expression to be considered is then

$$\text{Range} \doteq V\overline{\eta/c} \int_{W_2}^{W_1} dW/\text{t.hp.} \qquad (3)$$

and if we define† "mean effective thrust horsepower" as

$$\text{m.e.t.hp.} = (W_1 - W_2) \Big/ \int_{W_2}^{W_1} dW/\text{t.hp.} \qquad (4)$$

then

$$\text{Range} \doteq V\overline{\eta/c}\,(W_1 - W_2)/\text{m.e.t.hp.} \qquad (5)$$

If the lift-drag polar can be approximated by a curve which divides the drag coefficient in two parts, one independent of the lift coefficient C_L and the other proportional to the square of the lift coefficient, *i.e.*, if

$$C_D = C_{D_p} + C_{D_i} \doteq f/S + C_L^2/\pi eAR \qquad (6)$$

over a sufficiently large spread of C_L, then the thrust horsepower required for level flight is

$$\text{t.hp.} = \text{t.hp.}_p + \text{t.hp.}_i \doteq 0.00000682 f\sigma V^3 + 0.332 W^2/eb^2\sigma V \qquad (7)$$

where C_{D_p} = parasite drag coefficient; C_{D_i} = induced drag coefficient; t.hp.$_p$ = t.hp. required to overcome parasite drag; t.hp.$_i$ = t.hp. required to overcome induced drag; f = equivalent parasite area[2], sq. ft.; S = total wing area, sq. ft.; e = airplane-wing efficiency factor;[2] $AR = b^2/S$ = aspect ratio; b = equivalent monoplane wing span, ft.; σ = air mass density ratio.

The integration in Eq. (4) is carried out by substitution for t.hp. in terms of W from Eq. (7), and the resulting m.e.t.hp., expressed in terms of t.hp.$_1$, the t.hp. required at speed V, altitude corresponding to density ratio σ, and initial gross weight W_1, is

$$\frac{\text{m.e.t.hp.}}{\text{t.hp.}_1} = \frac{[1 - (W_2/W_1)]R_1^{1/3}/(1 + R_1)}{\arctan\{[1 - (W_2/W_1)]R_1^{1/3}/[1 + R_1(W_2/W_1)]\}} \qquad (8)$$

It has been convenient here to use a generalization of a ratio introduced by Diehl,[3] who uses a parameter R_m in range calculations which is the ratio, at maximum speed, of induced drag to parasite drag. He uses also a plain R to indicate the drag ratio at 100 m.p.h. at sea-level. In this paper R should be understood to be generalized to apply to any speed V being considered. In Eq. (8) R_1 indicates the value of R at the initial weight W_1, and at the cruising speed V and air density ratio σ being considered.

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† The "mean effective thrust horsepower" is so termed to eliminate confusion with mean t.hp., which would be

$$\overline{\text{t.hp.}} = \int_{W_2}^{W_1} \text{t.hp. } dW/(W_1 - W_2).$$

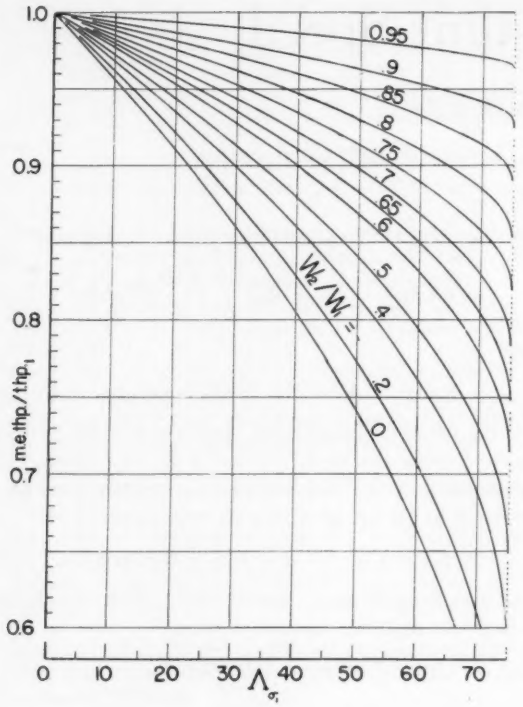

FIG. 1.

Eq. (8) is shown graphically in Fig. 1. The abscissa, however, is not R_1, but Λ_{σ_1}, the value of Λ_σ for V, σ and W_1. Λ_σ is a generalization[4] of Oswald's basic performance parameter Λ to hold for any altitude and any speed, and is defined by

$$\left.\begin{aligned} \Lambda_\sigma &= W^2 f^{1/3}/\sigma^{2/3} e b^2 \text{ t.hp.}^{4/3} \\ \text{whence} \quad \Lambda_{\sigma_1} &= W_1^2 f^{1/3}/\sigma^{2/3} e b^2 \text{ t.hp.}_1{}^{4/3} \end{aligned}\right\} \tag{9}$$

Λ_{σ_1} and R_1 are related as follows:

$$\left.\begin{aligned} R_1 &= \Gamma_{\sigma_1}/(1 - \Gamma_{\sigma_1}) \\ \Lambda_{\sigma_1} &= 158.9 \Gamma_{\sigma_1}(1 - \Gamma_{\sigma_1})^{1/3} \end{aligned}\right\} \tag{10}$$

where Γ_{σ_1} is the ratio of induced drag to total drag at V, σ and W_1, *i.e.*, $\Gamma_{\sigma_1} = C_{D_{i1}}/C_{D_1} = \text{t.hp.}_{i1}/\text{t.hp.}_1$.

Fig. 1 shows the variation of m.e.t.hp./t.hp.$_1$ with Λ_{σ_1} for various values of W_2/W_1. Since W_2/W_1 is known in a given design problem, whereas Λ_{σ_1} must be estimated, a large working chart would be better plotted against W_2/W_1 and values of Λ_{σ_1} interpolated. Such a working chart could eliminate the high and low values of W_2/W_1, but would need to cover almost the complete range of values of Λ_{σ_1}. (Note that if $\Lambda_{\sigma_1} = 63.1$, V is then the speed for maximum L/D ratio at σ and W_1; on the other hand, if V is the maximum speed at σ and W_1, then Λ_{σ_1} may be quite small.)

An analytical expression is easily determined for the error introduced in using the thrust horsepower required at the average weight. It is found that use of t.hp.$_{\overline{W}}$ is a surprisingly good approximation for what have so far been usual values of W_2/W_1 and of Λ_{σ_1}. For extreme values of these parameters, the error becomes as much as 6% of m.e.t.hp.

It should be noted that Eq. (5), with m.e.t.hp. as defined in Eq. (4), can be transformed to equations identical with Diehl's[3] Eq. (311) and Eq. (312). The simplicity of Eq. (5) and the ease of obtaining m.e.t.hp. from the curves makes more precise range estimates at constant velocity less foreboding than do Diehl's Eq. (311) and Eq. (312).

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The Design of High-Speed Military Airplanes

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Introduction

THE DESIGN of present military airplanes is changing so fast that any discussion of the subject is likely to be very far behind the latest trends by the time it is published. At the beginning of the war this was particularly true. It seemed that in a period of a few weeks many factors which should have been apparent in military designs made themselves known. The lack of armor, fuel tank protection and ample armament required hasty revision of many outstanding types. After a short while, the air forces of the major military powers could be evaluated in terms of their outstanding shortcomings or merits.

The success of Great Britain's eight-gun fighter and power turrets was one of the dominating factors in turning back Hitler's attacks last fall. The foresight of the British in providing such fundamentally sound design elements cannot be too highly praised.

Perhaps the outstanding features of the German air force were its great use of the dive bomber, the highly coordinated tactical coordination between the air forces and ground troops and its successful use of fuel injection engines and protected fuel tanks.

As the war developed, it became apparent that our own air force had called the turn very successfullly with its development of the long-range four-engined bomber, which had been greatly criticized in Europe, the turbo supercharger, and its prize bomb sight. While all countries had in some measure at least experimented with most of the above features, and in some cases had carried the respective developments (such as dive bombing) to high degrees of perfection, it is believed that the items indicated above were the outstanding elements in the air forces of the countries named from a point of view of technical development.

When the early war experience showed up the shortcomings of many service airplanes in armor, tank protection and armament, a great hue and cry was raised in the United States bemoaning the fact that our aircraft were so far behind the European types in these factors. Properly analyzing the facts in the case, however, shows that, with the exceptions noted above, all of the major air forces were very nearly on a par in being caught without proper protection and hitting power.

In this paper, the discussion will be limited to a discussion of design trends of the high-speed military fighter, although many of the statements made apply equally well to all types of high-speed airplanes. In considering the fighter airplane, its basic purpose in intercepting and shooting down bombers must be considered. The success of the fighter in stopping daylight raids has been well known, but developments in tactics of night bombing have made very difficult the problems of night interception and fighting. A few simple figures can be used to point out some difficulties which are encountered when it is desired to stop a bombing raid.

A modern bomber flying 300 m.p.h. at 25,000 feet can stop both engines, feather the propellers and glide practically noiselessly for a distance of 45 miles before obtaining an altitude of 5000 feet, eleven minutes after starting its glide. This gives the bomber an excellent chance for surprise raids, particularly at night when they cannot be seen. Naturally, the organization of ground forces, black-outs and various defense means are being developed to overcome the temporary advantage of the bomber, but the technical problems involved are very great. Considering the fact that our present bomber flying at high altitude with full power can add 100 m.p.h. to its speed in a shallow dive of only $5^1/_2$ degrees, it can be seen that sustained high speeds for long distances are obtainable when starting from high altitude. The new bombers of even cleaner aerodynamic design, having high speeds approaching 400 m.p.h., can also add 100 m.p.h. at about the same angle of power glide. Starting at 30,000 feet in such a glide, the bomber covers 50 miles in a period of six minutes while still having an altitude of 10,000 feet at the end of this time.

A pursuit plane ready to take to the air five minutes after the bombing plane has been sighted must have tremendous performance to climb to altitude and catch the bomber at any point on its attack. Under conditions of this nature, a speed advantage of 50 m.p.h. seems to be the absolute minimum acceptable for the fighter airplane over the bomber.

One of the obvious answers to such difficulties, of course, is the continual patrol of fighter squadrons over all vulnerable points at all times of the day and night. Naturally, the number of aircraft required for such patrol becomes very large.

Factors Effecting Design

The hardest problem which the designer of military airplanes must face is the proper evaluation of these

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fundamental factors in their relative importance, one to the other:

(a) Performance
(b) Armament
(c) Ease of production
(d) Maneuverability
(e) Time required for development
(f) Flight qualities
(g) Cost

All aircraft design is made up of compromises between these considerations and the success of the basic design depends on which of the above factors have been favored without too great a loss in the other items. The element of time is of paramount importance under wartime conditions because the manufacturer, or, more properly, the nation, cannot afford to gamble too far on new and untried design features for fear that trouble experienced in their development may make his airplane obsolete and waste his production facilities at a time when the greatest output is required. It is better to have 1000 four-hundred m.p.h. airplanes in service than to have the finest four hundred and fifty m.p.h. design in the world on paper struggling with problems of design and production. Only after sound fundamental research and the highest type of engineering can the manufacturer afford to use some of the newer developments now on the horizon of progress.

Performance

Of the various items of airplane performance, the most important for the pursuit or interceptor type is speed. Present tendencies in the war indicate that high speed should be obtained at as high an altitude as possible. At the time of this writing, service pursuit craft now being used have high speeds of 330 to 390 m.p.h. between 15,000 and 23,000 feet. The types now in early production stages will reach 370 to 450 m.p.h. between 20,000 and 30,000 feet. In order to obtain such velocities, continual research must be directed on all phases of airplane engine and propeller design. Fig. 1 shows a comparison between the major performance characteristics of a typical present-day fighter and the improvements which might logically be expected for service airplanes in the near future.

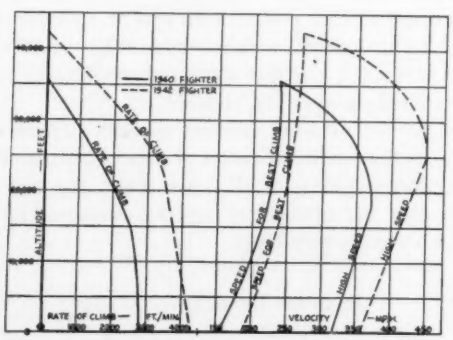

FIG. 1. Comparative performance of typical present fighter and probable 1942 type.

A rapid tendency toward higher wing loadings and lower power loadings can be noted, brought about by the desire for higher speed and greater load carrying ability of the fighter aircraft. Table 1 shows the effect of varying the wing loading only on the basic pursuit function of interception. A measure of the effectiveness of the fighter airplane is the distance it can cover in a given time from take-off, when considered for a certain operating altitude. For wing loading values now in sight, it will be seen that improved performance results with the higher wing loadings. The second factor which forces an increase in wing loading is the great increase in the weight of armament, armor and power plants required for keeping in line with the bomber development. The effect of wing loading on maneuverability seems to be gradually getting lost, due to the limitations of the pilot in withstanding the load factors so easy to develop at high speeds.

TABLE 1

All airplanes designed to carry same engines, armor, armament and crew. Gross weights vary to account for difference in wing and tail weight. Stability is constant.

Wing Loading	Time to 20,000 Ft.	Max. Speed, 20,000 Ft.	Distance from Take-off Point after 30 Minutes (20,000-Ft. Alt.)
30 lbs./sq.ft.	6.15 min.	400 m.p.h.	180 miles
45 lbs./sq.ft.	5.88 min.	425 m.p.h.	192 miles
60 lbs./sq.ft.	5.92 min.	445 m.p.h.	202 miles

Drag

The relation of airplane drag and thrust horsepower for airplanes in the 400 m.p.h. class require that extremely careful attention be given to the reduction of air resistance if the airplane size and power are to remain within the bounds of practicality.

The designer must continually attempt to make the smallest possible airplane for a given power and be sure that the drag of every element is as low as can be obtained without sacrificing stability, control or maximum lift. Figs. 2 and 3 show the energy breakdown for a typical modern fighter airplane in the higher performance brackets. The drag of the various elements of the airplane has been reduced greatly from values in existence several years ago. There is still a possibility of a substantial reduction in the skin frictional resistance, roughness and cooling drag. By using airfoils with shapes which extend the laminar flow area the drag can be reduced considerably, but these wing sec-

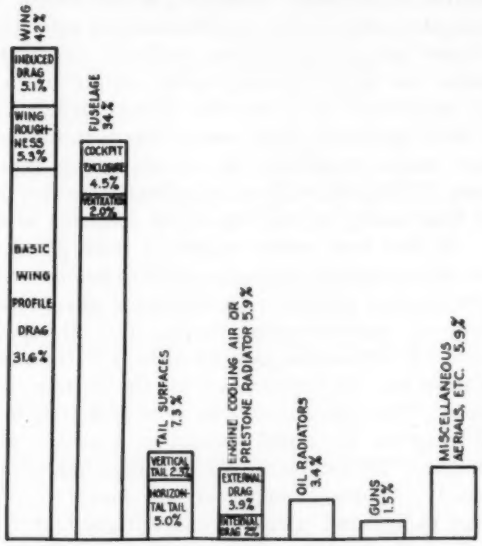

FIG. 2. Drag breakdown for a typical single engine fighter airplane having performance shown in Fig. 1. Total airplane drag = 100%. Total thrust power at high speed = 1041 hp. (see Fig. 3).

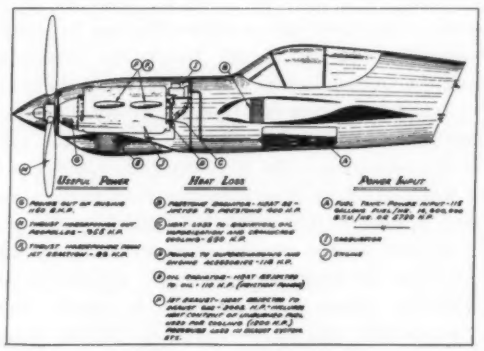

FIG. 3. Energy balance for typical military engine—operating at maximum power output.

tions must be considered from the point of view of manufacturing difficulties, maximum lift and the effect of roughness which is inevitably encountered in service when flying off rocky fields or through hail storms. There is an undeveloped field requiring much additional research on this problem which seems to show much promise.

There has been little change in the heat rejection for cooling of our modern engines for several years. As engine powers increase and operating altitudes go up, the difficulties of providing sufficient cooling assume great proportions. While the projected area of modern engines has decreased, the size of the radiators required for cooling has increased to the point where the design of a modern military fighter attains the condition where practically all equipment items and structure must be developed around the power plant. Even though the projected area of liquid-cooled engines has been held down very well for the power obtained, the fuselage size encountered in the installations of such engines is generally disappointing to everyone concerned because of the problem of installation of superchargers, intercoolers, oil radiators, Prestone radiators and landing gear wheels of a size to take the weight of the engine involved.

The classic argument of the lower drag of the liquid-cooled engine *versus* the air-cooled type for a given power output seems no nearer solution now than it ever has been. The designers of the liquid-cooled engines enjoy drawing pictures of their liquid-cooled engines with fine pointed noses and faired shapes, forgetting to look back a few feet on the airplane and finding under the fuselage, or in the wings, the same type of air entrance that must be used for cooling the air-cooled engine. There is no theoretical reason and little practical evidence to show that lower drag per horsepower can be obtained with a liquid-cooled installation than can be obtained with the air-cooled type. In making a large number of comparative designs, it has also been noted that very little difference in the projected area or fuselage drag results for the two installations. As regards compressibility effects on cowling of the air-cooled type, it is as easy to avoid critical pressure distributions on a cowling nose as it is on a radiator duct.

When considering the drag of the single engine fighter airplane, the remarkable aerodynamic cleanliness of the present type can be shown by comparison with an equivalent flat plate area. The total drag for such an airplane at any given speed is no more than that of a flat plate 22 inches square. The poor designs, however, run up to twice this drag figure. A good twin-engine fighter can be represented in drag by a flat plate 27 inches square.

WEIGHT

It is very distressing to the designer, the manufacturer, the Air Force, and finally to the tax payer, that the trend in military aircraft design is toward higher weight, greater complication and great complexity. It has often been stated that designs should be simplified, weights reduced and the design considered primarily from a manufacturing point of view. Serious efforts have been made by all parties concerned to try to bring about such a trend, but it has been impossible to eliminate any of the fixed equipment or power plant items to obtain greater simplicity. A second best airplane cannot be tolerated in actual combat. Any means of transportation having a speed of 400 m.p.h. is inevitably complicated.

It is interesting to make a weight breakdown of a typical modern pursuit airplane. The percentage of weight over which the airplane designer has control is only 40 per cent of the airplane gross weight. Twenty-eight per cent of the airplane gross weight consists of

stressed material working to a very high unit stress under the design load factors (which are of the nature of 12). The remaining weight is made up of power plant and propeller, radio, instruments, tires, wheels, armament, fuel, oil and miscellaneous equipment. These are all items over which the airplane designer has no weight control. Figs. 4 and 5 illustrate typical breakdowns.

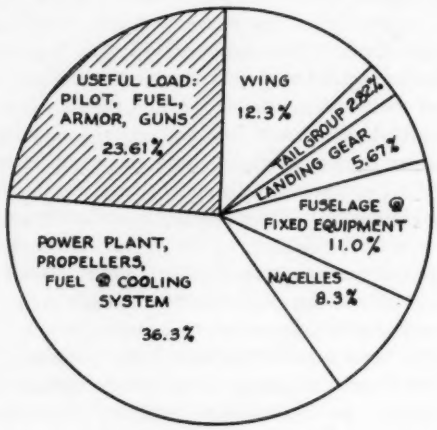

FIG. 4. Gross weight breakdown of two-engine fighter.

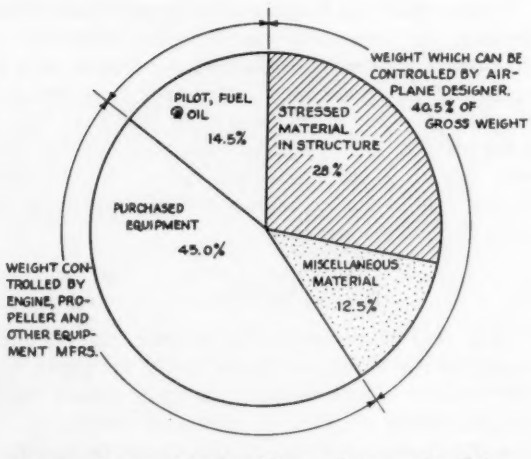

FIG. 5. Gross weight breakdown of two-engine fighter, showing percentage of weight within designer's control.

As airplane speeds increase, the percentage of weight in the power plant increases also. Developments in the present war have shown the need for increased equipment rather than a reduction of what is now used. It can be seen that future military airplanes tend to become larger and more expensive as the performance is pushed up.

ENGINES

The modern high altitude military engine is a marvelous machine when compared to other types of power generating equipment. Restricted in size, weight, fuel consumption and cooling, it represents the most highly developed single unit of the airplane. In order to visualize the thermodynamic cycle and obtain an energy breakdown for a modern power plant, Fig. 3 has been prepared. The energy input at maximum power output at altitude for an engine of approximately 1150 hp. amounts to 115 gallons of fuel per hour. The total energy of the fuel input amounts to 5730 hp. Of this total power supply, a large percentage must be rejected for cooling in order to maintain satisfactory engine stresses. An additional amount is required for supercharging, cooling the oil, etc., as shown. A substantial amount of heat is rejected by radiation and oil vaporization on the bottom of the pistons. This loss amounts to more heat than is carried away by the liquid coolant or from the finned cylinders. The largest source of energy loss is in the exhaust. At maximum power, the fuel-air ratio required to limit detonation is substantially higher than the theoretical ratio for maximum power. The excess fuel provided for this condition is present in the exhaust in the partially burned state and is normally wasted. The exhaust gas leaving the cylinders has a temperature of roughly 1700°F., and the cracking pressure of exhaust valves allows the gas to escape at pressures around 70 to 90 lbs./sq.in. From these figures, the tremendous power loss can be understood.

A part of this power can be regained by the use of jet exhausts or a turbo supercharger, but at the present state of development the recovery percentage has been very low. A great deal of research is being directed toward finding better ways for regaining this power. The exhaust energy can be used not only by kinetic means, but the heat energy can be applied to steam systems for power development. As speeds increase and propulsive efficiency decreases due to compressibility effects, it is not out of the realm of possibility to consider engines which furnish their power for takeoff and climb to a propeller which can be feathered for level flight and, in the latter condition, the total engine power output being used by highly developed jets for propulsion.

In order to maintain its power output at altitude, the engine is supercharged one of several ways. The most usual type of supercharger consists in a centrifugal blower geared directly to the engine with a fixed gear ratio. This same unit has been developed whereby a gear shift has been incorporated and two gear ratios are available, one for low altitude and one for high. The most advanced type of gear-driven supercharger incorporates two superchargers with an intercooling stage between the two. The third means of maintaining altitude power involves the use of the turbo supercharger. This unit is driven by exhaust gas and maintains high power to higher altitudes than any other type of supercharging in use. Fig. 7 shows a comparison between the powers available at altitude when the

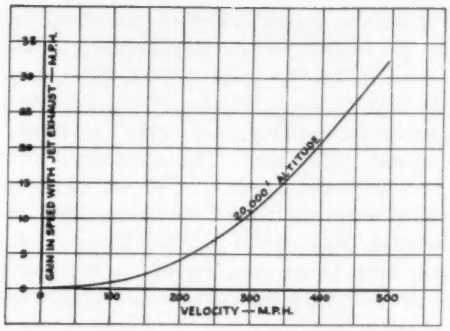

FIG. 6. Increase in speed available from simple jet reaction at 20,000 feet altitude.

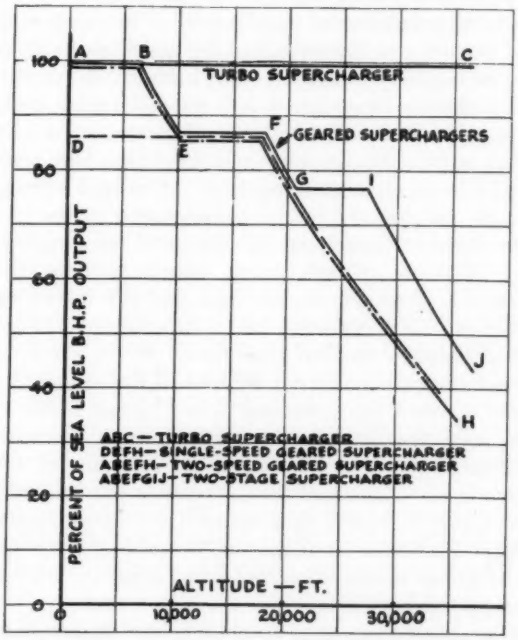

FIG. 7. Comparison of supercharger types.

above means are used for supercharging. It is apparent that the turbo supercharger is very superior to other types at high altitudes. Difficulties in cooling, high installation drag and weight, as well as complexity in control, have all hindered the adoption of turbos. Intensive research has now solved the above difficulties to the point where turbo superchargers are reliable and easily handled items of equipment. The United States Army Air Corps deserves great credit for their fostering this fine development.

Most superchargers now in use in military aircraft are of the single- and two-speed type. The closest approach to overall turbo performances, however, can be obtained using the two-speed, two-stage supercharger engine equipped with jet exhaust. Inasmuch as the turbo derives its power from the engine exhaust by maintaining a constant back pressure in the exhaust system it has not as yet been found possible to obtain any substantial jet reaction in combination with the turbo supercharger at high altitude. At low altitudes there is no reason why jet reaction cannot be obtained with the turbo operating at partial capacity. Fig. 8 shows the comparative net thrust horsepower available in an airplane equipped with the turbo supercharger in one case and the two-stage engine with jet exhaust in the other. For altitudes above approximately 20,000 feet the turbo supercharger has better performance than its rival, but at lower altitudes, present indications are that the two-stage engine with jets is superior. (See Fig. 6.)

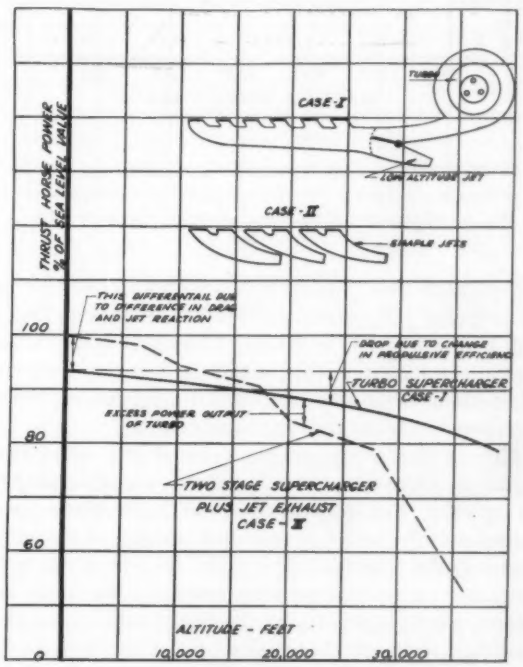

FIG. 8. Approximate performance comparison of turbo supercharger and two-stage geared supercharger plus simple jet exhaust.

PROPELLERS

It is becoming increasingly difficult to maintain good propulsive efficiency on military airplanes at high altitudes. Fig. 9* shows a summary of computed propulsive efficiency at high speeds using the best high-speed wind tunnel data available to date. While this information seems to be very pessimistic, the trend established is, no doubt, correct. The curves shown are derived from wind tunnel tests on airfoil sections subjected to two dimensional flow. It is known that compressibility effects are less for flow in three dimensions similar to that which exists around propeller tips

* *Propeller Design Problems of High Speed Airplanes*, H. B. Dickinson, presented at the Airplane Design session, Ninth Annual Meeting, I.Ae.S., New York, January 31, 1941.

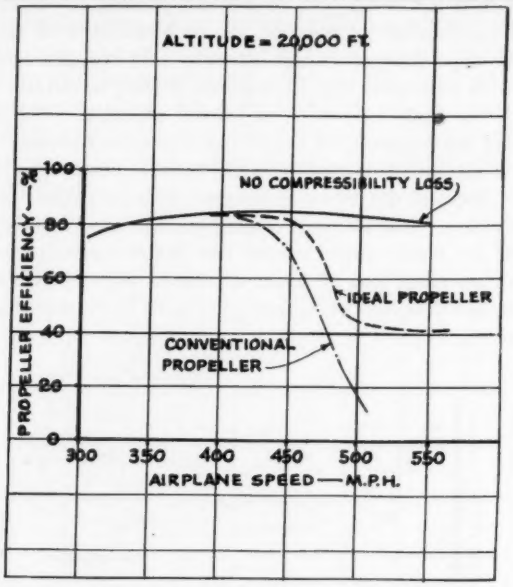

FIG. 9. Computed variation of propeller efficiency with velocity.

and perhaps over considerable sections of the rest of the propeller. Research on propellers operating at very high forward speeds is entirely lacking in spite of the continual requests on the part of the industry for such information from the various Government Agencies equipped to obtain this vital information.

There is increasing need for a gear shift to allow two engine-propeller gear ratios in order to maintain proper tip speeds. The requirements for high blade area necessitates the use of an increased number of propeller blades so the four-bladed propeller is coming into use. Tests run on counter-rotating propellers have not indicated substantial gains for the dual rotation propeller compared to the single rotation unit of the same blade area. The use of pusher propellers, desirable for many points of view when considering drag, is being retarded by practical considerations such as the problem of landing gear wheels throwing rocks back into the blades, flutter conditions developing when wing flaps are extended, pilot escape in emergency and the problem of ground clearance as it affects landing gear length.

At the speeds encountered with fighter airplanes, the compressibility shock wave tends to develop as quickly at the propeller hub where the sections are thicker as it does at the propeller tip where the speeds are greater. To avoid this condition, propeller hub fairings have been developed. These have not as yet proved very satisfactory except for cooling in radial engines, as the practical limitations on the chord of the fairings prevent airfoil sections of the proper thickness ratio from being used. The problem of pitch distribution also must be carefully investigated before any substantial gain in speed will result from their use. Several tests on conventional cuffs have indicated a definite speed loss when placed on an airplane in the higher performance brackets. We can expect to see a definite trend toward wider propeller blades, perhaps obtaining the aspect ratios used on the high-speed Supermarine racers in the Schneider Cup Races several years ago. The application of wide blades or wide fairings near the hub will be limited by the mechanical capabilities of the mechanisms used to obtain variable blade angles and feathering. Considerable study must be directed toward obtaining propeller sections allowing higher critical compressibility speeds.

ARMAMENT

The modern fighter airplane must carry a substantial weight of armament to be effective. The number of guns carried and the weight of bullets or shells fired must be carefully evaluated. For instance, should the fighter airplane be equipped with ten 0.30 caliber guns or an equivalent weight but a smaller number of 0.50 caliber guns? Are one cannon and four machine guns as effective as eight machine guns? This type of controversy requires the careful consideration of the airplane designer throughout the design of the airplane. The continual struggle being waged between the armament protection on one side and the increasing caliber of guns on the other makes for continual change in the design as the war progresses. It is possible to protect the pilot and vital sections of the aircraft by means of armor plate between $1/4$ and $1/2$ inch in thickness from the fire of 0.30 caliber guns. Likewise, bulletproof glass $1\,1/2$ inches thick can protect the pilot's face. The use of the 0.50 caliber gun and cannon, however, makes such protection entirely inadequate. Fuel and oil tanks can be protected by the use of rubber composition tanks, but again the cannon changes this condition.

When considering the two-place fighter, flexible guns mounted in a power turret are usually considered. At the speeds of our modern airplanes, it is impossible for a man to swing large guns against an airstream, so electrical or hydraulic means must be used to relieve him of this duty. The installation of power turrets on a high-speed airplane is very difficult, particularly when airplanes pressurized for high altitude are considered.

STRUCTURE

One of the amazing things about the modern high-speed airplane is the fact that they are so strong with so many cut-outs made in the basic structure to account for the installation of radiators, guns, landing lights, landing gear, flaps, fuel tanks, intercoolers and a host of other miscellaneous items. Fig. 10 shows a typical example of how much space there is left for structure in the wing of a fighter after providing for the above items. The difficulty of providing rigidity for flutter prevention is easily visualized.

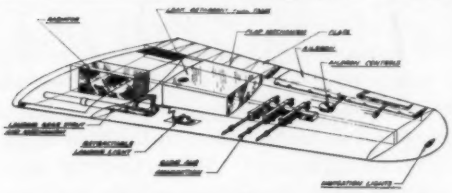

FIG. 10. Typical example of items enclosed by wing structure on a fighter airplane.

One of the outstanding problems of structural design at the present time is the provision of adequate strength to take the loads developed in high-speed dives where large portions of the airplane are subjected to compressibility phenomena. In many cases the normal air loads existing are quadrupled when the shock wave forms over some portion of the surface.

Increased landing speeds, greater sinking velocities and heavier engine installations add to the problems of the structural engineer who is caught by increasing loads on one side and the requirements for thinner wing sections and smaller fuselages on the other.

ONE ENGINE OR TWO?

The relative advantages of the twin engine airplane *versus* the single engine type can be summed up rather easily by making two comparisons. For the first case consider two airplanes, one a single engine type with an engine of a given power and the other a twin engine type having two power plants identical to the single engine airplane. The twin engine airplane will weigh from 75 to 85 per cent more than the single engine type and therefore will have the advantage in drag per horsepower and power loading. For this condition the relative advantages of the two airplanes are indicated below.

Twin Engine Airplane Advantages	Single Engine Airplane Advantages
1. Higher speed.	1. Cheaper to build.
2. Greater range at given speed.	2. Easier to service.
3. Safety with one engine out of commission.	3. Greater maneuverability due to smaller size.
4. Ability to use safely higher wing loadings.	
5. Higher rate of climb.	
6. Higher ceiling.	
7. Generally better pilot vision (new types of single engine airplanes may eliminate this advantage).	

The second comparison representing an opposite extreme develops when a single engine airplane having a total installed power equal to the twin engine airplane is considered. In this case, the single engine airplane will weigh about 80 per cent as much as the twin engine type, will have higher performance, greater range, higher ceiling and climb than the twin engine type. It will be almost as expensive to build, however, and as difficult to maintain as the twin engine airplane of the same total power. The above conclusions assume equal excellence of design. The twin engine type will naturally maintain a higher safety, resulting in smaller losses during wartime use.

It is not logical to expect that at equal stages of development we shall see designs developed along the line where the total power of the twin engine airplane is no greater than that of a single engine type, so the comparison between the one- and two-engine fighter will always lie between case 1 and case 2 above. This is true up to the point where the twin engine fighter obtains such velocities that it encounters a large drag increase due to compressibility effects. The single engine type can then overtake its rival in spite of the power advantage of the two-engine type. When this condition prevails, the single engine type is subjected to the same drag increase as its rival. Further speed comparisons cannot be made until some answer is found to the basic problem common to both types. Previous experience has always shown that there is a way to remove seemingly insuperable barriers when the proper time comes.

COMPRESSIBILITY EFFECTS

Throughout this paper, casual references to compressibility effects have been made. Without going into any details, the term will be explained briefly. Whenever the flow speed of the air around any part of the airplane equals the speed of sound, that is, the rate of propagation of a pressure wave in air, a shock wave is formed causing an entirely different type of air flow from the type encountered at subsonic velocities. When such a shock wave forms, a considerable amount of energy is dissipated as heat and the drag of the body increases considerably. Naturally, the first part of the airplane to encounter compressibility effects is the propeller, which is subjected to the rotational velocity as well as the forward speed of the airplane. A continual struggle is being engaged in by the aerodynamicists to find shapes which keep the critical velocity of all the component parts of the airplane as high as possible. The limiting speed of many of our present wing sections, for instance, is about 450 m.p.h. We have every reason to expect that newer developments will put this speed higher. It so happens that every development which tends to increase the compressibility speed also reduces the drag at lower velocities, so definite benefits will be obtained for all performance conditions.

PILOT LIMITATIONS

With the airplane advancing technically at such a rapid rate, consideration must be given to the pilot

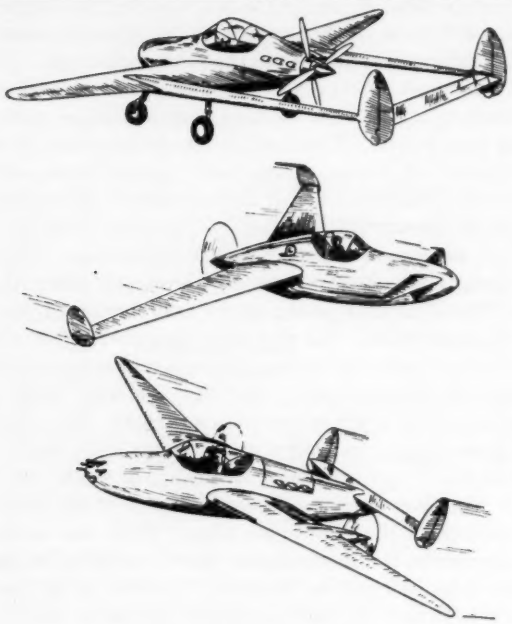

FIG. 11. Probable future types of fighter airplanes.

and his physical limitations. At high speeds in straight flight, of course, there is absolutely no problem involved, but the moment that a turn is made, high accelerations placed on the pilot cause him considerable discomfort and result in rapid fatigue. With the critical altitude of the airplane being increased, the problem of the pressure comfort of the pilot is an important one.

Using pure oxygen, a pilot is fairly efficient at altitudes of 30,000 feet. Above this height, his reflexes become very slow and his strength reduces rapidly. During tests on the Lockheed twin engine interceptor airplane it has been found advisable to have the pilot saturate himself with oxygen on the ground before making rapid climbs to high altitudes. This procedure is hardly suitable for pursuit operation in war time, so it seems that the development of either pressure cabin airplanes or pressurized suits will be necessary to allow the pilot to obtain the maximum performance from the more advanced type of airplanes. This further complication adds to all the difficulties outlined above in this paper and makes the airplane more vulnerable, heavier and more expensive.

It is known that when a pilot is in a prone condition he can withstand approximately twice the acceleration that he can while sitting upright and some study has been made on designs in which the prone position is used for fighting.

The value of increased performance of fighter airplanes must be carefully evaluated against pilot reaction and his physical well-being.

CONCLUSIONS

Fig. 11 shows several promising trends in airplane design which bear out the discussion given above. In spite of the fact that the pusher propeller has many problems, it seems fairly certain on the basis of present data that it should be used. Before they can be widely used, the problem of the pilot leaving the airplane with parachute must be considered. With a rotating propeller behind him, it is absolutely impossible to leave the airplane safely, so means for stopping it in a given position, dropping it or throwing the pilot clear, must be incorporated. Fig. 12 shows variation of engine installation weight with critical altitude for maximum power. (Including propellers, accessories, cooling and oiling system, motor mounts, cowling, etc.) The Lockheed P-38, front view, is shown in Fig. 13. Note the small projected area of the different component parts.

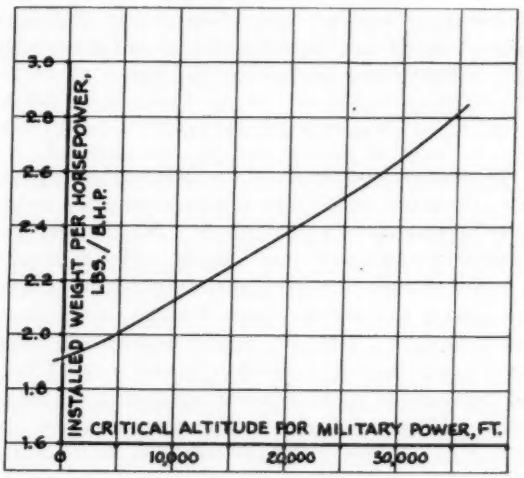

FIG. 12.

FIG. 13.

The military fighter airplane is inevitably becoming larger, more powerful, more costly and complicated. It is useless to discuss limitations to speed, as all such previous discussions have proved in error to date. The most useful outcome from the vast amount of research being put on military types will be its application to commercial models when the war is over. Under the pressure of present times, development is being carried on which, under peace-time conditions, would not have been available for years.

On the Statistical Theory of Turbulence

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ABSTRACT

In the following section a method is given of determining the distribution function of turbulent velocities. The distribution function $P(u)$ of a component turbulent velocity $u(t)$ is defined as the probability that u shall have the value between u and $u + du$. The part $P_e(u)$ of $P(u)$ which is an even function of u can be determined by first measuring the mean with respect to time of $\cos zu(t)$ for all values of z. If this function be denoted by $\psi(z)$, then $P_e(u)$ is given by

$$P_e(u) = \frac{1}{\pi}\int_0^\infty \cos uz\ \psi(z)dz$$

In a similar fashion, the part $P_o(u)$ of $P(u)$ which is odd in u can be determined by measuring $\varphi(z)$, which is $\overline{\sin [zu(t)]}$, and obtaining the sine transform of $\varphi(z)$:

$$P_o(u) = \frac{1}{\pi}\int_0^\infty \sin uz\ \varphi(z)dz$$

In the section on A Method of Determining the Auto-Correlation Coefficient of Turbulent Velocities a new method is given of determining the auto-correlation coefficient $R(\tau)$ of $u(t)$. This method depends on the possibility of measuring a function $U(t, \theta)$, which is the mean value of $u(t)$ over an interval of time 2θ

$$U(t, \theta) = \frac{1}{2\theta}\int_{t-\theta}^{t+\theta} u(t)dt$$

$U(t, \theta)$ has, like $u(t)$, a zero mean value and the mean of its square $\overline{U^2(t, \theta)}$ is equal to $\overline{u^2(t)}$ at $\theta = 0$ and decreases with θ. If now $r(\theta)$ is defined by

$$r(\theta) \equiv \frac{\overline{U^2(t, \theta)}}{\overline{u^2(t)}}$$

then the auto-correlation coefficient $R(\tau)$ of $u(t)$ is given by

$$R(2\theta) = \frac{1}{2}\frac{d^2}{d\theta^2}[\theta^2 r(\theta)]$$

THE DISTRIBUTION FUNCTION OF TURBULENT VELOCITIES

THE DISTRIBUTION function of turbulent velocities was measured by Townend[1] and Simmons and Salter[2] who obtained a Gaussian distribution for the case of uniform turbulence. In the following an alternative method of determining the distribution function is suggested, which involves measurements which are more akin to current techniques in turbulence studies. We shall assume, in the first instance that the distribution function $P(u)$ is even in u and shall begin with a discussion of

$$Q(a) = 2\int_0^a P(u)du$$

which gives the probability that u lies between $-a$ and $+a$. We shall define $Q(a)$ in the following way. We consider the trace of $u(t)$ as a function of t and choose a portion of this curve covering a large interval of time T, during which $u(t)$ goes through many oscillations. Over a portion T_1 of T $|u(t)|$ is less than a, and $Q(u)$ is defined as the limit of the ratio (T_1/T) as T is increased indefinitely. Our problem is to determine this limit by operating on $u(t)$. To do this we use Dirichlet's discontinuous factor

$$\varphi(u, a) = \frac{2}{\pi}\int_0^\infty \frac{\sin az}{z}\cos zudz = 1,\ u^2 < a^2$$
$$= 1/2,\ u^2 = a^2 \quad (1)$$
$$= 0,\ u^2 > a^2$$

The integral $\int_{-T}^{T} \varphi[u(t), a]dt$, therefore gives the fraction $2T_1$ of the total time interval $2T$ over which $u^2 < a^2$. It follows that

$$Q(a) = \operatorname*{Lim}_{T\to\infty}\frac{1}{2T}\int_{-T}^{T}\varphi[u(t), a]dt \quad (2)$$

If we substitute in (1) and change the order of integration, we obtain

$$Q(a) = \operatorname*{Lim}_{T\to\infty}\frac{2}{\pi}\int_0^\infty \frac{\sin az}{z}dz\,\frac{1}{2T}\int_{-T}^{T}\cos[zu(t)]\ dt =$$
$$\frac{2}{\pi}\int_0^\infty \frac{\sin az}{z}\overline{\cos[zu(t)]}\ dz \quad (3)$$

where the bar indicates the mean with respect to t. Assuming that the differentiation is permissible, we get from (3)

$$P(a) = \frac{1}{2}\frac{d}{da}Q(a) = \frac{1}{\pi}\int_0^\infty \cos az\ \psi(z)dz \quad (4)$$

where we have written $\psi(z)$ for $\overline{\cos[zu(t)]}$. The latter function has an interesting development in terms of the moments of u. If we expand $\cos[zu(t)]$ into a power series and then take the mean, we obtain

$$\psi(z) = 1 - \frac{z^2}{2!}\overline{u^2} + \frac{z^4}{4!}\overline{u^4} - \frac{z^6}{6!}\overline{u^6} + \ldots \quad (5)$$

so that $\psi(z)$ determines the even moments of u and with that, necessarily, the even part of the distribution function. As an example we might taken a Gaussian distribution for which

$$P(u) = \frac{1}{\sqrt{2\pi\,\overline{u^2}}}e^{-(u^2/2\overline{u^2})} \quad (6)$$

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* Research Associate, Dept. of Geology.

It follows from (4) that

$$\psi(z) = \frac{2}{\sqrt{2\pi\,\overline{u^2}}}\int_0^\infty \cos\ uz\ e^{-(u^2/2\overline{u^2})}\,du = e^{-(z^2\overline{u^2}/2)}$$

$$= 1 - \frac{\overline{z^2u^2}}{2} + \frac{1}{8}z^4(\overline{u^2})^2 - \frac{1}{48}z^6(\overline{u^2})^6 + \ldots \qquad (7)$$

By comparing with (5) we see that for a Gaussian distribution

$$\overline{u^4} = 3(\overline{u^2})^2,\ \overline{u^6} = 15(\overline{u^2})^3, \text{ etc.} \qquad (8)$$

relations that can be obtained directly from the distribution function. It is clear that the corresponding relation to (4) for the odd part of the function is

$$P_0(u) = \frac{1}{\pi}\int_0^\infty \sin\ uz\ \cdot\ \overline{\sin\,[zu(t)]}dz \qquad (9)$$

A Method of Determining the Auto-Correlation Coefficient of Turbulent Velocities

In his paper, *The spectrum of turbulence*,[3] Taylor has shown that with $I_1(k)$ and $I_2(k)$ defined by

$$I_1(k) = \frac{1}{\pi}\int_{-T}^{T} u(t)\cos\ kt\ dt \qquad (10)$$

$$I_2(k) = \frac{1}{\pi}\int_{-T}^{T} u(t)\sin\ kt\ dt\ ,\ k = 2\pi n \qquad (11)$$

the portion $F(n)$ of $u^2(t)$ in the frequency range n and $n + dn$ is given by

$$F(n) = 2\pi^2 \operatorname*{Lim}_{T\to\infty}\left(\frac{I_1^2 + I_2^2}{T}\right) \qquad (12)$$

It was also shown that the correlation coefficient $R(\tau)$ of $u(t)$ is given by

$$R(\tau) = \int_0^\infty F(n)\cos\ 2\pi n\tau\ dn \qquad (13)$$

Now if $u(t)$ were of integrable square and the limits in (10) extended from $-\infty$ to $+\infty$, an inversion of (10) would yield

$$u_e(t) = \frac{1}{2}\int_{-\infty}^{\infty} I_1(k)\cos\ kt\ dk \qquad (14)$$

Let $U(t, \theta)$ be defined by

$$U_e(t, \theta) = \frac{1}{2\theta}\int_{t-\theta}^{t+\theta} u_e(\tau)d\tau \qquad (15)$$

then it follows from (14) that

$$U_e(t, \theta) = \frac{1}{4\theta}\int_{-\infty}^{\infty} I_1(k)dk\int_{t-\theta}^{t+\theta}\cos\ k\tau\ d\tau$$

$$= \frac{1}{2\theta}\int_{-\infty}^{\infty} I_1(k)\cos\ kt\ \frac{\sin\ k\theta}{k}\ dk \qquad (16)$$

In a similar way we obtain

$$U_0(t, \theta) = \frac{1}{2\theta}\int_{-\infty}^{\infty} I_2(k)\sin\ kt\ \frac{\sin\ k\theta}{k}\ dk \qquad (17)$$

It follows that to the Fourier transforms $I_1(k)$ and $I_2(k)$ of $u(t)$ there correspond the functions $I_1(k)\ \dfrac{\sin\ k\theta}{k\theta}$ and $I_2(k)\ \dfrac{\sin\ k\theta}{k\theta}$ for $U(t, \theta)$.[4] Hence, by analogy with (12), we can write for the frequency distribution G_n of $U^2(t, \theta)$

$$G_n = F_n\frac{\sin^2(2\pi n\theta)}{(2\pi n\theta)^2} \qquad (18)$$

This relation is to be interpreted in the sense that in a frequency range between n and $n + dn$, where the amplitude of $u^2(t)$ is $\overline{u^2}F(n)dn$, the corresponding amplitude of $U^2(t, \theta)$ is $\overline{u^2}F(n)\ \dfrac{\sin^2(2\pi n\theta)}{(2\pi n\theta)^2}\ dn$.

Since, by definition, $\displaystyle\int_0^\infty F(n)dn = 1$, we conclude that

$$r(\theta) = \frac{\overline{U^2(t, \theta)}}{\overline{u^2(t)}} = \int_0^\infty F(n)\ \frac{\sin^2\,(2\pi n\theta)}{(2\pi n\theta)^2}dn \qquad (19)$$

It follows from (19) that

$$\frac{d^2}{d\theta^2}\left[\theta^2 r(\theta)\right] = 2\int_0^\infty F(n)\cos\ 4\pi\ n\theta dn = 2R(2\theta)^5 \qquad (20)$$

References

[1] Townend, H. C. H., Proc. Roy. Soc., A Vol. 145, page 180, 1934.

[2] Simmons, L. F. G., and Salter, C., Proc. Roy. Soc., A Vol. 145, page 212, 1934.

[3] Taylor, G. I., *The Spectrum of Turbulence*, Proc. Roy. Soc., A Vol. 164, page 478, 1938.

[4] A rigorous treatment of the highly involved argument leading to the derivation of Eqs. (12) and (13) will be found in N. Wiener's paper on *Generalized Harmonic Analysis*, Acta Mathematica, Vol. 55, page 118, 1930.

[5] For a treatment of some of the mathematical subtleties involved in the argument of the section on The Distribution Function of Turbulent Velocities, see Levy, P., *Calcul des Probabilitités*, Gauthier-Villars, Paris, 1925. It was pointed out to the author by Dr. Hugh L. Dryden that the problem of the section on A Method of Determining the Auto-Correlation Coefficient of Turbulent Velocities was treated by a different method by Skramstead in N.A.C.A. Technical Report No. 581, 1937. By the method used in this section one can easily prove that the auto-correlation coefficient of the derivative of $u(t)$ is proportional to the second derivative of $R(\tau)$.

Institute Notes

Pacific Aeronautical Library Opened

On October 1 the Institute opened the Pacific Aeronautical Library at 6715 Hollywood Boulevard, Hollywood, California. This will provide in a central location convenient to members in the Southern California region one of the largest collections on the West Coast of books, reports and periodicals on aeronautics.

Establishment of the new library was made possible by a grant of $10,000 from The Paul Kollsman Fund of $50,000 which was set up in 1940 by Paul Kollsman and the Square D Company to provide an aeronautical lending library service for the Institute. Many of the aircraft companies on the West Coast are participating in the operation of the Pacific Aeronautical Library.

This service library for aeronautical research will be available to the qualified public for reading privileges, with books circulating to the libraries of the various Southern California aircraft companies. Source material includes aerodynamic and structural research reports as well as books on drafting, production methods, history and allied sciences.

Over 1000 volumes will be available immediately, but in addition the library will be able to secure any aeronautical book on loan from the 5000 books in The Paul Kollsman Library and 180,000 engineering books in the Engineering Society's Library in New York. The Pacific Aeronautical Library is an outgrowth of a plan presented to Major Lester D. Gardner, Executive Vice-President of the Institute, by a committee made up of librarians from the various aircraft companies in the West Coast district. The Institute of the Aeronautical Sciences, through this newly established library, is furnishing books, periodicals and pamphlet material which will be circulated to the participating aircraft companies, to supplement their engineering libraries.

E. W. Robischon, who has been aeronautics librarian at the California Institute of Technology for the past five years, is in charge of the library, which will be open from 9 until 5 o'clock daily, with evening hours to be established in the near future.

Members of the Western Aeronautical Libraries Committee, who were instrumental in obtaining the cooperation of the Institute of the Aeronautical Sciences in establishing the Pacific Aeronautical Library, include Clement S. Skrabak, Lockheed Aircraft Corporation, Chairman; Patricia Kelly, Douglas Aircraft Co., Secretary; Tonea Hamilton, Hughes Aircraft Co.; Mona Harmer, Northrop Aircraft, Inc.; E. W. Robischon, Aeronautics Library, California Institute of Technology; E. W. Hjorth, Vega Airplane Co.; and Emma Quigley, member of the Special Libraries Association, National Defense Committee.

R. V. Southwell Wright Brothers Lecturer on December 17, 1941

Professor Richard Vynne Southwell, F.I.Ae.S., F.R.Ae.S., of the University of Oxford, England, has accepted the invitation of the Institute to give the Wright Brothers Lecture this year. Endowed in 1937 by the Vernon Lynch Fund, this will be the fifth of these lectures commemorating the anniversary of the first flights made by the Wright brothers at Kitty Hawk in 1903. It will be held at Columbia University, New York, on the evening of Wednesday, December 17. The subject will be "New Pathways in Aeronautical Theory."

Professor Southwell received his M.A. degree from Trinity College, Cambridge University in 1912. From 1915 to 1918 he was engaged in the design of non-rigid airships at the Royal Naval Air Station in Kent and from 1918 to 1919 was in charge of full-scale aerodynamic research and testing of airplanes at the Royal Aircraft Establishment, Farnborough. He was Superintendent of the Aerodynamics Department, National Physical Laboratory from 1920 to 1925. In 1925 he returned to the faculty of Cambridge University, where he had been a Fellow and lecturer from 1912 to 1920, and was lecturer on Mathematics until 1929. Since then he has been Professor of Engineering Science at the University of Oxford, from which he also has an M.A. degree.

Professor Southwell is the author of research papers on elasticity, hydrodynamics, theory of structures and other aeronautical applications. He was awarded the R.38 Memorial Prize in 1926 for a paper on stress calculation in rigid airship hulls and was co-winner of the same prize in 1930 for a continuation of this study. He is a Member of the Institution of Mechanical Engineers and a Fellow of the Royal Society.

Daniel Guggenheim Medal for 1941 to Juan T. Trippe

The Daniel Guggenheim Medal Board of Award announced last month the selection of Juan T. Trippe, M.I.Ae.S., President of Pan American Airways System, to receive the Medal for 1941. The Daniel Guggenheim Medal, established in 1928 as an award for notable achievement in the advancement of aeronautics, is awarded to Mr. Trippe "for the development and successful operation of oceanic air transport." It will be presented to him at the Honors Night Dinner of the Institute to be held in New York on January 27.

One of the pioneers of American transport aviation, Juan T. Trippe is the founder of the Pan American Airways System. Mr. Trippe began his career in aviation during World War I when he enlisted in the Naval Air Service in which he became an Ensign. After the Armistice, he returned to Yale to complete his course in the Sheffield Scientific School, from which he was graduated in 1920. While at Yale, he organized the first student flying club in the country.

Soon after graduation he helped to organize Colonial Airways, the first commercial air transport company in the United States. Awarded Domestic Air Mail Contract Number 1, Colonial Airways operated a route between New York and New England under his management.

In 1928 Mr. Trippe was successful in organizing a group of former pilots who pledged the financing necessary to undertake the first American international aerial transport system. Under the name of Pan American Airways he established the first United States foreign route between Key West and Havana late in 1927.

Mr. Trippe is Director of and Vice Chairman for the United States of the Air Transport Commission of the International Chamber of Commerce, President of the International Air Traffic Association, the directing body for international air transport throughout the world, a member of the Executive Committee of the Business Advisory Council for the Department of Commerce; and a member of the Advisory Board of the Institute.

Recipients of the Daniel Guggenheim Medal are chosen by elected representatives of the Institute of the Aeronautical Sciences, American Society of Mechanical Engineers, Society of Automotive Engineers, United Engineering Trustees and the previous recipients of the Medal, who are Life Members of the Board of Award. Those who received the award in previous years are: Orville Wright, 1929; Ludwig Prandtl, 1930; F. W. Lanchester, 1931; Juan de la Cierva, 1932; J. C. Hunsaker, 1933; William E. Boeing, 1934; William F. Durand, 1935; George W. Lewis, 1936; Hugo Eckener, 1937; Alfred H. R. Fedden, 1938; Donald W. Douglas, 1939; and Glenn L. Martin, 1940.

Chairmen of Technical Sessions Tenth Annual Meeting

The Council of the Institute has appointed the following Chairmen for the Technical Sessions of the Tenth Annual Meeting to be held at Columbia University, New York City, January 28–30, 1942:

Aerodynamics—Dr. Arnold M. Kuethe, University of Michigan
Airplane Design—Fred E. Weick, Engineering and Research Corp.
Air Transport—Charles Froesch, Eastern Air Lines
Materials—C. F. Nagel, Aluminum Company of America
Meteorology—C. F. Brooks, Blue Hill Observatory, Harvard University
Physiological Problems—Col. Arnold D. Tuttle, United Airlines Transport Corp.
Power Plants—R. N. Du Bois, Packard Motor Car Company
Radio and Instruments—W. A. Reichel, Pioneer Instrument Div., Bendix Aviation Corp.
Rotating-Wing Aircraft—Ralph H. McClarren, The Franklin Institute
Structures—E. E. Lundquist, Langley Memorial Aeronautical Laboratory, N.A.C.A.

All members wishing to submit papers for any of these sessions of the Annual Meeting are asked to send them directly to the Institute. Titles and outlines of papers or preferably the complete paper should be received by the end of this month. They will be forwarded to the Chairmen of the sessions concerned, and authors will be notified on or before November 15 as to whether their papers can be included in a session of the Meeting. All papers submitted will be considered by the Editorial Board for publication in the Journal. While a paper should be planned to take only twenty to twenty-five minutes for oral presentation at the meeting in written form they may have a maximum length of from 10,000 to 15,000 words. The form to be followed in preparing papers is described in "Suggestions for Contributors" which may be secured by writing to the Institute's New York Office.

Gates Elected Honorary Member

Artemus L. Gates, recently appointed Assistant Secretary of the Navy for Air, has been elected to Honorary Membership in the Institute.

Mr. Gates was born in Cedar Rapids, Iowa, November 3, 1895. In 1917 he joined the U.S. Naval Air Service and was a flying officer with the Northern Bombardment Group, which served on raiding missions along the German coast in cooperation with French and British Forces. In July 1918 he was made commanding officer, with the rank of Lieutenant, of the U.S. Naval Air Station at Dunkerque, France. Later that year, while flying with a naval squadron, he was brought down by anti-aircraft fire behind the German lines. He managed to burn his plane before being captured by the Germans, who held him a prisoner until the end of the war. He was awarded the American Distinguished Service Medal and the British Distinguished Flying Cross.

Since World War I, Mr. Gates has been active in New York financial circles. He is President of the New York Trust Company and a member of the Board of Directors of Pan American Airways System.

As Assistant Secretary of the Navy for Air, a post which has been vacant since 1932, Mr. Gates will act as liaison between the Navy Bureau of Aeronautics and aircraft manufacturers in handling material procurement in the greatly expanded defense program.

Warner a Benefactor Member

At its last meeting the council designated as a Benefactor Member of the Institute, Edward P. Warner, F.I.Ae.S., Vice-Chairman of the Civil Aeronautics Board. Mr. Warner recently gave to the Institute a valuable collection of over two hundred aeronautical books and a large number of reports, periodicals and miscellaneous aeronautical papers. This gift is particularly useful to the Institute's Archives, not only because of the new items it has added to the reference collection, but also because of the duplicates that can now be made available to The Kollsman Lending Library and the new Pacific Aeronautical Library.

Charles Froesch Elected to Council

At the meeting of the Council of the Institute on September 10, Ralph S. Damon, A.F.I.Ae.S., resigned from membership on the Council. Formerly with American Airlines, he is now President of Republic Aviation Corporation.

To fill his unexpired term until the next Annual Meeting, Charles Froesch, A.F.I.Ae.S., Chief Engineer of Eastern Air Lines, was elected. His membership on the Council provides the desired representation of an airline on the governing board of the Institute.

News of Institute Members

Federal Loan Administrator Jesse Jones has appointed a committee of aeronautical experts to design a cargo plane especially suited for use in South America. Head of this project, which is a part of the American Republics Aviation Division of the Defense Supplies Corporation, is **Reed M. Chambers,** M.I.Ae.S., who is on part time leave from the Vice Presidency of U.S. Aviation Underwriters to serve with the Division. On the Committee are: **John W. Crowley, Jr.,** A.F.I.Ae.S., Head of the Flight Research Section of the N.A.C.A.; **A. A. Priester,** F.I.Ae.S., Vice President and Chief Engineer, Pan American Airways; **William B. Stout,** F.I.Ae.S., President, Stout Engineering Laboratories. Allan Lockheed, aircraft designer, Eric Nelson, formerly of Boeing Aircraft Co., and Luther Harris, Vice President—Maintenance, Pennsylvania-Central Airlines complete the membership of the committee.

Donald L. Bower, M.I.Ae.S., Engineer with Wright Aeronautical Corp., has been assigned to active duty as a 1st Lieutenant in the Central Air Corps Procurement District of the U.S. Army at Detroit where he is Assistant to the Production Engineering Officer.

Griffith Brewer, Hon.M.I.Ae.S., Hon.F.R.Ae.S., begins this month his second term as President of the Royal Aeronautical Society of Great Britain, having been re-elected to this office last summer.

Jean H. DuBuque, M.I.Ae.S., previously C.A.A. Supervisor of Ground School Training in the Civilian Pilot Training Service at San Antonio, Texas, has joined Aero Insurance Underwriters as Manager of their Chicago branch.

The Senate has confirmed the appointment of **Lt. Gen. Delos C. Emmons,** Hon.M.I.Ae.S., to the permanent rank of Major General from the date of March 1, 1939. He now holds the temporary rank of Lieutenant General and is the Commanding General of the Air Force Combat Command.

William C. Gould, M.I.Ae.S., has left Brewster Export Corporation, where he was Assistant to the President, and is now Assistant Service Manager of the Allison Division of General Motors Corp. in Indianapolis.

Dr. Theodore von Kármán, Hon.F.I.Ae.S., Director of the Guggenheim Aeronautical Laboratory, California Institute of

Technology, has been appointed by Northrop Aircraft, Inc., as consultant to the company on aerodynamics and structural engineering.

William P. Kennedy, M.I.Ae.S., who has been with the Airplane Division of the Curtiss-Wright Corp. for the last two years in the sales department at the Buffalo Plant, has been named Assistant to the General Manager of the plant now under construction at Columbus, Ohio.

Dr. Arnold M. Kuethe, A.F.I.Ae.S., recently an Aeronautical Engineer at the Langley Memorial Aeronautical Laboratory, N.A.C.A., has joined the faculty of the University of Michigan as Associate Professor in the Department of Aeronautical Engineering.

John D. Peace, Jr., A.F.I.Ae.S., formerly Manager of Aeronautical Sales for Sperry Products, Inc., is now with United Aircraft Products, Inc., in charge of sales for Dayton and the Pacific Coast.

E. Robert Reff, M.I.Ae.S., formerly with Northrop Aircraft, Inc., is now Structures Engineer on the staff of the Boeing Aircraft Co.

Eugene Reynolds, Technical Member, has left Spartan Aircraft Company where he was an Aeronautical Engineer to join the Engineering Department of Wichita Division, Boeing Aircraft Co. (formerly Stearman Aircraft Co.).

Waldon R. Rhoads, Technical Member, formerly Design Engineer with the Airplane Division (St. Louis Plant), Curtiss-Wright Corp., is now with Vega Airplane Company as Design Engineer for Controls.

Frederick K. Teichmann, A.F.I.Ae.S., Assistant Professor in New York University's Guggenheim School of Aeronautics, which is now under the direction of **Charles H. Colvin,** F.I.Ae.S. has been appointed Acting Head of the Department of Aeronautical Engineering in charge of academic matters.

Robert S. Wolfsohn, Technical Member, has left his previous position as Junior Aeronautical Engineer with the N.A.C.A. at Langley Field to join the staff of the National Bureau of Standards in Washington.

President Roosevelt appointed on September 23 the United States National Commission of the Permanent American Aeronautical Commission, with Thomas Burke, Chief of the State Department Division of Communications, as Chairman and including in its membership Richard E. Elwell, Chief Counsel of the C.A.A., and Samuel E. Gates, International Counselor of the C.A.B., who constitute an Executive Committee. The other appointees to the American Commission, all of whom are Institute members are:

Reed M. Chambers, M.I.Ae.S., Vice President, U.S.Aviation Underwriters.

John C. Cooper, Jr., M.I.Ae.S., Vice President, Pan American Airways System.

Harold R. Harris, A.F.I.Ae.S., Vice President, Pan American-Grace Airways, Inc.

Dr. J. C. Hunsaker, Hon.F.I.Ae.S., Coordinator of Research and Development, Navy Dept.

Col. John H. Jouett, M.I.Ae.S., President, Aeronautical Chamber of Commerce of America.

Dr. George W. Lewis, Hon.F.I.Ae.S., Director of Aeronautical Research, N.A.C.A.

Dr. Ross A. McFarland, M.I.Ae.S., Assistant Professor, Fatigue Laboratory, Harvard University.

Captain Denis Mulligan, M.I.Ae.S., United States Army Air Corps, Wright Field.

T. P. Wright, F.I.Ae.S., Assistant Chief, Aircraft Branch, Office of Production Management.

This Commission was established pursuant to agreements reached at the Pan American Conference at Lima in 1937 to provide a permanent inter-American body to work toward the unification and codification of international private and public air law and to coordinate aeronautical technical activities in the Western Hemisphere.

Personnel Opportunities

The Personnel Bureau serves individual members, as well as organizations seeking to employ aeronautical specialists. Any member or organization may have requirements listed without charge.

Wanted

The Parker Appliance Company is now augmenting its key personnel in Management, Engineering and Production. An invitation is extended to men ready for advancement to submit evidence of executive ability in their respective fields. Though not wishing to interfere with the production of other defense industries, we would appreciate the applications of men whose talents are not being utilized to the utmost in their present capacity. Address replies to The Parker Appliance Company, Cleveland, Ohio.

Wanted

Meteorological Observers. An examination has been announced by the United States Civil Service Commission to secure persons for the position of Assistant Observer in Meteorology at $1620 a year in the Weather Bureau of the Department of Commerce. Positions of Junior Observer in Meteorology at $1440 a year and Under Observer in Meteorology at $1260 a year may also be filled from the employment registers established as a result of this examination.

Observers in meteorology take, record, chart and compute meteorological and radiosonde observations and disseminate weather reports and warnings for the benefit of agriculture, commerce, aviation and other interests. To be admitted to the written test for these positions, applicants must have completed either 2 years of college study including mathematics or physics, or 2 years of full-time paid experience as a meteorological observer.

Applications must be on file at the Commissions' Washington Office *not later than October 23, 1941.* Application forms may be obtained at any first or second-class post office.

Available

Aircraft Production and Engineering Executive. Qualified by 17 years military and commercial type aircraft experience, B.S.E. (Ae.E.) degree. Chief Engineer. Factory Manager. Address reply to Box 141, Institute of the Aeronautical Sciences.

New Corporate Members

The following companies joined the Institute as Corporate Members last month:

Aircraft Accessories Corporation, Burbank, Calif., manufacturers of such hydraulic devices for aircraft as selector valves, cylinders, pumps, regulators, relief valves, etc. Its subsidiary **Aircraft Accessories Corporation of Missouri,** at Kansas City, Kansas, manufactures propeller governor test units, radio transmitters and radio accessories.

Firestone Aviation Products Company, Akron, Ohio, recently formed aviation subsidiary of the Firestone Tire & Rubber Company, which handles the tires, tubes, rubber tubing, rubber hose and other products of the company used in aviation.

The Parker Appliance Company, Cleveland, Ohio, makers of tube couplings, fittings and valves for airplane hydraulic systems, bending equipment and other aircraft specialties.

John A. Roebling's Sons Company, Trenton, N. J., manufacturers of wire rope for the past one hundred years, whose numerous products now include airplane control cord, control cable terminals and fittings, galvanized aircraft wire, ignition and lighting circuit cables.

United-Carr Fastener Corporation, Cambridge, Mass., makers of the "Dot" line of fasteners, including airplane cowl fasteners and snap fasteners, and also special nuts and screws, metal stampings, laminated plastics and radio parts for aeronautical uses.

Changes in Membership

The following changes in the membership of the Institute have occurred since the publication of the previous issue of the Journal.

Transferred to Associate Fellow Grade

Percival, Edgar Wikner, F.R.Ae.S.; Founder & Director, Percival Aircraft Ltd., England.

Stewart, Earle, B.S. in M.E.; Asst. Design Engineer, Landing Gears, The Glenn L. Martin Co.

Elected to MEMBER Grade

Chapman, Everett, M.S.; President, Lukenweld, Inc. (See Who's Who in Eng.)

Kinsey, Robert S., B.S. in C.E.; Field Engineer, Wright Aeronautical Corp.

McFadgen, Henry Marion, Vice-Pres. Engineering, Jacobs Aircraft Engine Co,

Martin, Harold Clifford, M.S.; Instructor in Aero. Engineering, New York University.

Maurenbrecher, Henri Anton, Test Pilot and Purchasing Officer, Netherlands Purchasing Commission.

Mines, Chester Eugene, B.S. in M.E.; Resident Chief Engineer, Jacobs Aircraft Engine Co.

Prince, David Chandler, M.S. in E.E.; Manager Commercial Engineering, General Electric Co.

Ramberg, Walter, D.Sc.; Senior Physicist, National Bureau of Standards.

Wolowicz, Chester Henry, M S.; Instructor in Aeronautics and Mechanics, Northeastern University.

Transferred to Member Grade

Klenke, William Henry, Jr., Asst. Sales Mgr. & Engineering Test Pilot, Vultee Aircraft, Inc.

Elected to Industrial Member Grade

Budds, Henry Harold, General Manager, Airplane Parts Div., Briggs Mfg. Co.

Jordanoff, Assen, B.S.; Aviation Author (See W.W. in A.).

Parker, John Scott, Jr., Chief, Ground School Section, Civil Aeronautics Administration.

Simon, S. Thomas, B.S. in M.E., J.D.; General Counsel's Office, Civil Aeronautics Board.

Elected to Technical Member Grade

Alexander, John Day, B.S. in Ae.E., J.D.; Aero. Engineer, Stinson Aircraft Div., Vultee Aircraft, Inc.

Branning, Ralph Ivan, Jr., Draftsman, Douglas Aircraft Co.

Burke, Woodward, B.S. in M.E.; Chief Test Pilot, Brewster Aeronautical Corp.

Chamberlain, Orville Virden, B.S. in M.E.; Asst. Instructor in Airplane Mechanics Course, U.S. Army Air Corps Technical School, Chanute Field.

Cohen, Eli Lawrence, B.S. in Ae.E.; Aero. Engineer, Brewster Aeronautical Corp.

Daniels, Walter, Jr., Aerodynamics Dept., North American Aviation, Inc.

Hutzler, Richard Hochschild, B.S. in E.; Engineer, Allied Aviation Corp.

McConnell, Robert A., B.S. in Phys.; Physicist, Naval Aircraft Factory.

Noltenmeier, Edward Herbert, Inspector, Brewster Aeronautical Corp.

Petresek, William August, M.S.; Electrical Engineer, American Airlines, Inc.

Pichenot, Lucien Francis, M.E.; President, Nigucol Aircraft.

Summers, Thomas Garner, A.A. in Ae.; Junior Test Engineer, Hughes Aircraft Co.

Necrology

Carl Wieselsberger

Prof. Dr. Carl Wieselsberger, F.I.Ae.S., died on April 26, 1941, after a prolonged illness, at Aachen, Germany.

Dr. Wieselsberger, who was among the original group of foreign scientists invited to membership in the Institute when it was founded, had been Professor of Aerodynamics and Director of the Aerodynamics Institute of the Aachen Technische Hochschule since 1930.

Born in Eberstahl, Bavaria, November 3, 1887, he studied in industrial and technical schools in Munich and received his degree as Doctor of Engineering from the Technische Hochschule of Munich in 1912, his dissertation being on the stability of aircraft. From 1912 to 1922, Dr. Wieselsberger was an assistant and collaborator of Dr. Ludwig Prandtl at the Aerodynamic Research Laboratory of Göttingen University, except for a period in 1914–15 when he volunteered for military service and received training as an airplane pilot. In 1922 he received a Doctorate of Philosophy from Göttingen.

For the next eight years he was Scientific Associate at the Aerodynamic Institute of the Imperial University of Tokyo and helped establish several aeronautical research laboratories in Japan. In 1930 he returned to Germany to direct aeronautical instruction and research at Aachen.

Dr. Wieselsberger was associated with that international group of scientists whose work on the equipment and methods for accurate aerodynamic measurements contributed so greatly to the development of the science. His reports on wind tunnel design and instruments, fluid mechanics, the effects of air flow at high velocities and other aerodynamic studies have been published in the proceedings of the three famous aeronautical laboratories with which he was associated and in numerous technical periodicals. He was a member of the German Academy for Aeronautical Research and the Lilienthal Society for Aeronautical Research. He had received the Service Cross for War Aid (German) and the Gold Cup awarded by the Emperor of Japan for his work at the Imperial University.